NASA Contractor Report 4072

Space Station Experiment Definition: Long-Term Cryogenic Fluid Storage

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FORWORD

This Final Report summarizes the technical effort conducted by Beech Aircraft Corporation, Boulder Division, under Contract No. NAS3-24661. The contract was administered by the Lewis Research Center of the National Aeronautics and Space Administration, Cleveland, Ohio. The study was performed from September 1985 to May 1986. The NASA-LeRC Project Manager was Mr. Myron Hill. The author also wishes to acknowledge the contributions of Mr. John C. Aydelott of NASA-LeRC.

A listing of the Beech Aircraft personnel who contributed to this study is presented below, including their primary areas of contribution:

D. H. Riemer	Program Manager
R. D. Scarlotti	Project Engineer
R. L. Jetley	Lead Engineer and Thermal/Fluid Analysis
J. E. Beach	Design
D. L. Rohs	Design
D. L. Barnett	Program Plan
W. F. Wildhaber	ROM Costing

In addition, Dr. Walter Unterberg of Rockwell International, Rocketdyne Division, provided information on Space Station resources and interfaces.

The data in this report are presented with the International System of Units as the primary units and English units as secondary units. All calculations were made in English units and converted to the international units.

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SUMMARY

This study presents the conceptual design of a Space Station Technology Development Mission (TDM) experiment to demonstrate and evaluate cryogenic fluid storage and transfer technologies. Cryogenic technologies required by future orbital systems, such as Orbital Transfer Vehicle refueling stations, were determined and critical technologies to be demonstrated by the experiment were chosen. The experiment will be deployed on the Initial Operational Capability (IOC) Space Station for a four year duration. It is modular in design, consisting of three phases to test the following technologies:

Phase I - Passive Thermal Technologies
Phase II - Fluid Transfer Technologies
Phase III - Active Refrigeration Technologies

Use of existing hardware was a primary consideration throughout the design effort. This resulted in recommendations to use several pieces of existing hardware (or their designs), including the Oxygen Thermal Test Article (OTTA) as a cryogen supply tank, and the Earth Limb Measurement Satellite (ELMS) tank as a receiver tank.

A conceptual design of the experiment was completed, including configuration sketches, fluid system schematics, equipment specifications, and Space Station resource and interface requirements. These Space Station requirements were documented utilizing the NASA Mission Requirements Data Base (MRDB) and Technology Development Advocacy Group (TDAG) forms. The information from these forms will be incorporated into the NASA Space Station data base, and will allow Phase C/D Space Station design efforts to be responsive to the needs of this TDM.

A preliminary evolutionary plan was developed defining the overall program schedule and Rough Order of Magnitude (ROM) costs required for experiment development and operation. This effort defined a twelve year development and flight plan, at a total cost of $94.3M (1986 dollars).

1.0 INTRODUCTION

This report presents the work performed under NASA Contract NAS3-24661 entitled "Space Station Experiment Definition: Long-Term Cryogenic Fluid Storage." The primary objective of this study was to develop a conceptual design for a Space Station Technology Development Mission (TDM) experiment that will demonstrate and evaluate the technologies required for long-term storage and transfer of cryogenic fluids in an orbital environment. Space Station resource and interface requirements were then defined utilizing the Technology Development Advocacy Group (TDAG) and the Mission Requirements Data Base (MRDB) forms. Early requirements definition allows design efforts conducted in Phase B and Phase C/D of the Space Station program to be responsive to these needs. The targeted time frame for flight of this experiment is the mid-1990's.

1.1 Background. Planning efforts are currently underway at NASA to establish mission guidelines and requirements for a Space Station which will be operational in the mid 1990's. Proposed missions have been solicited from the science, technology, and commercial communities, and a preliminary data base has been established which defines the mission requirements. These TDM experiments are conducted with the support of Space Station and utilize long durations in the space environment to develop, test and evaluate advanced technologies for earth and space-based applications. Approximately 70 TDMs have been identified to date covering a broad range of technologies and disciplines and share the following characteristics:

1. Space Station is essential for the accomplishment of experimental objectives. Unique requirements may include long durations in space, availability of power, or availability of large spatial areas.

2. The technology is appropriate for the 1991-2000 time frame. The experiments are aimed at projected future needs and capability beyond the Initial Operation Capability (IOC) Space Station.

Such a projected future need is the deployment and maintenance of Orbital Transfer Vehicles (OTVs) from Space Station. These OTVs will utilize high specific impulse cryogenic engines. Plans for the growth Space Station include an OTV servicing and refueling facility. The technologies required for such a facility need further development and on-orbit demonstration prior to deployment. In addition to OTV refueling, liquid cryogens will be required for satellite servicing, life support systems, rapid quench thermal control, and general cooling of science and technology experiments. Storage tanks, with optimized insulation systems to minimize boiloff, must be large enough to store thousands of kilograms of cryogens such as liquid oxygen (LO_2) and liquid hydrogen (LH_2). Smaller quantities of liquid nitrogen (LN_2) and liquid helium (LHe) may also be used for long durations.

The Long-Term Cryogenic Fluid Storage Experiment (LTCFSE) is a TDM proposed by the NASA Lewis Research Center (LeRC) to demonstrate the technologies needed to satisfy these requirements.

1.2 Related Programs. Numerous programs are currently underway to develop technologies that will be demonstrated in the LTCFSE experiment. Development programs investigated during the study are listed in Table 1-1. Of the programs listed, the Cryogenic Fluid Management Flight Experiment (CFMFE) is the only experiment currently funded to include flight testing; all others include ground development only. CFMFE is a reusable test bed designed to be carried into orbit and demonstrated in the Shuttle cargo bay. The experiment hardware is configured to provide low-g verification of fluid and thermal models of cryogenic fluid storage and transfer processes. CFMFE will be used to demonstrate several critical technologies, such as no-vent tank fill, low-g quantity gaging and liquid acquisition. Since the experiment will be based in the Shuttle cargo bay, the tests will be conducted within the relatively short duration of less than one week. The objective of the LTCFSE is to extend the CFMFE technologies and provide the versatility to demonstrate additional technologies outside the scope of CFMFE.

Table I-I. RELATED TECHNOLOGY DEVELOPMENT PROGRAMS.

PROGRAM	SPONSORING AGENCY	TIME FRAME
Cryogenic Fluid Mgmt Flight Experiment	NASA-LeRC	1983 - 1993
Zero-G Quantity Gaging	NASA-JSC	1985 - 1987
Oxford Stirling Cycle Cooler	NASA-GSFC	1980 - 1989
Long Duration Exposure Flight Experiment	NASA-LaRC	1984 - 1990
Passive Orbital Disconnect Strut	NASA-ARC	1981 - 1990
Thick Multi-Layer Insulation	AFRPL	1986 - 1989
Long-Term Cryogenic Storage Facility Demonstration Program	NASA-MSFC	1985 - 1986
Cryo Cooler	AFWAL	1965 - present
Multi-Stage Magnetic Refrigerator	AFWAL	1982 - present
Metal Hydride Test Bed	NASA-MSFC	1986 - 1987
Sorption Compressor Refrigeration System	AFWAL	1986 - 1990
Compact Cryogenic Feed System	AFRPL	1986 - 1989

1.3 Scope of Effort. The LTCFSE study technical effort consisted of five tasks, as shown in Table I-II. The end result of these tasks was a conceptual design of the LTCFSE, along with the preliminary costs and schedule required to complete development, deployment, and on-orbit testing. Each task is described below.

Table I-II. TECHNICAL TASK BREAKDOWN.

Task I	Identification of Critical Technologies
Task II	Determination of Experimental Requirements
Task III	Documentation of Experimental Requirements
Task IV	Detailed Conceptual Equipment Design
Task V	Preliminary Evolutionary Plan

1.3.1 Task I - Identification of Critical Technologies. The objective of Task I was to identify critical technologies to be included in the experiment and to define an experiment plan to demonstrate and evaluate these technologies. Requirements for future orbital cryogenic systems were defined and compared to projected 1990 technology development levels. Technologies were chosen to be included in the experiment based on these requirements and the 1990 development levels. A preliminary experiment plan to demonstrate these technologies was developed. This plan was time-phased, so technologies that are both compatible and at similar stages of development will be tested simultaneously.

1.3.2 Task II - Determination of Experimental Requirements. The objective of Task II was to produce a conceptual design of the experiment to a level that allowed the requirements of Space Station resources to be documented and entered into the NASA Space Station data base. Before the conceptual design was begun, the restrictions imposed by system interfaces and use of existing hardware in the experiment were investigated. The intent of investigating the possible use of existing hardware was to minimize experiment development time and cost. The conceptual design consisted of a design description, including configuration sketches and equipment lists and Space Station resource and interface requirements.

1.3.3 Task III - Documentation of Resource Requirements. The experiment interface and resource requirements defined in Task II were documented in Task III, utilizing the MRDB and TDAG forms. These forms were delivered to NASA and will be entered into the Space Station data base.

1.3.4 Task IV - Detailed Conceptual Design. The objective of Task IV was to produce a detailed conceptual design of the experiment based on the Task II conceptual design. Detailed equipment sketches and system schematics were produced and control, contamination, safety and interface issues were investigated. In addition, a location on the Space Station for the experiment was chosen.

1.3.5 Task V - Preliminary Evolutionary Plan. The objective of Task V was to develop a program plan for the LTCFSE experiment. This program plan included a Work Breakdown Structure, Rough-Order-of-Magnitude costs by fiscal year and an overall program schedule.

2.0 RESULTS

The results of Tasks I through V are presented in detail in the following sections. Each subsection outlines the results of a particular task and includes a detailed description of the task's objective and approach.

2.1 Task I - Identification of Critical Technologies. The objective of Task I was to identify the cryogenic technologies that should be incorporated into the LTCFSE design and to define a preliminary experiment program to demonstrate and evaluate these technologies. The approach utilized to achieve this objective is presented in Figure 2-1. Potential technologies required by future orbital cryogenic systems were first identified. The projected 1990 development level of each of these technologies was estimated by reviewing the current 1985 State-of-the-Art (SOA) of each of these technologies, reviewing any pre-1990 development programs relating to these technologies and using this information to project the 1990 SOA. Critical technologies to be included in the experiment were then selected from the initial list of potential technologies based on the benefit and development level of each technology and the need for on-orbit demonstration. A preliminary experiment program was defined, separating the technologies into compatible groups and a scaling analysis was performed to determine an approximate experiment size. A test plan that time-phased the testing of these compatible technology groups based on development level was then prepared.

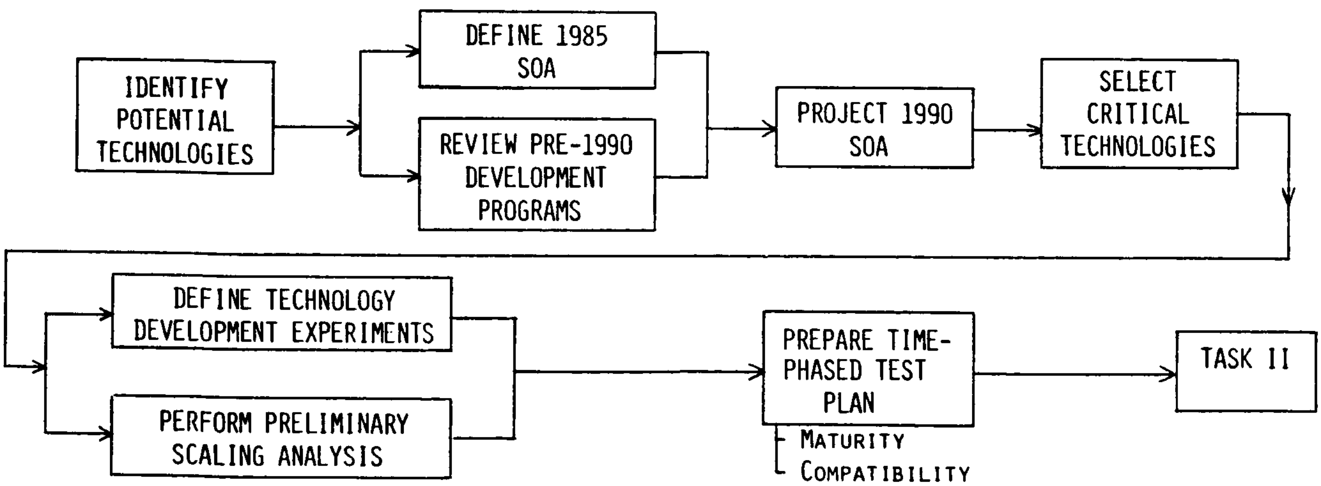


Figure 2-1. TASK I APPROACH.

2.1.1 Identify Potential Technologies. The first step within Task I was to identify any potential cryogenic technologies that may be included in the experiment. Potential technologies are those required by future orbital cryogenic systems, with particular emphasis on technologies needed for systems to be deployed in the late 1990's. Such applications are numerous and include OTV and satellite servicing, life support systems, rapid quench thermal control and instrument and sensor cooling. OTV servicing was emphasized in this study, as this application provides the firmest requirement for these cryogenic technologies in the late 1990's.

Space-based OTV operation will require three separate systems: a resupply tanker to deliver cryogen to the Space Station, a Space Station tank farm to store the cryogen and refuel the OTVs and the OTV itself. Future requirements of these systems are listed in Tables 2-I and 2-II. Table 2-I lists specific technology hardware required under three categories: Passive Thermal, Fluid Transfer and Active Refrigeration. Passive thermal technologies include those items that are utilized to reduce heat leak to cryogenic fluids. Fluid transfer technologies are those required for zero-g transfer of cryogenic fluids. Active refrigeration includes technologies required for an effective long-lifetime system that provides on-orbit refrigeration or reliquefaction of cryogenic fluids.

Table 2-II presents technology issues that must be addressed in the design and operation of these future systems. Each issue listed does not necessarily correspond to a piece of hardware, although in several cases, such as micrometeroid protection, the hardware must be designed to accommodate this phenomenon. Table 2-II is divided into three categories: Environmental Phenomena, Fluid Management and On-Orbit Logistics. Environmental phenomena are those phenomena peculiar to the orbital environment that may affect system performance. Examples of such phenomena are the effects of long-term fluid stratification and degradation of thermal coating optical properties. The second category, Fluid Management, refers to techniques and operations performed during fluid transfer and storage. On-Orbit Logistics are operational issues that will be encountered during utilization of these systems.

Table 2-I. FUTURE SYSTEM REQUIREMENTS - TECHNOLOGY HARDWARE.

HARDWARE CATEGORY	FUTURE REQUIREMENTS		
	RESUPPLY TANKER	OTV	SPACE STATION TANK FARM
PASSIVE THERMAL			
Dual Stage Support			X
Para-Ortho Conversion			X
Thick Multi-Layer Insulation			X
Thermodynamic Vent System	X	X	X
Thermal Coatings		X	X
Soft Outer Shell		X	X
Hard Outer Shell	X		
FLUID TRANSFER			
Capillary Acquisition	X	X	X
Low-G Quantity Gaging	X	X	X
Mass Flow Meters	X		X
Low Heat Leak Valves	X	X	X
Low Heat Leak Transfer Lines	X		X
Cryogenic Disconnects	X	X	X
External Pressurization	X	X	X
ACTIVE REFRIGERATOR			
Long Lifetime Refrigerator			X
Reliquefaction			X
Cryogenic Heat Exchanger			X
Refrigerator to Space Station Thermal Bus Heat Exchanger			X

Table 2-II. FUTURE SYSTEM REQUIREMENTS - TECHNOLOGY ISSUES.

ISSUE CATEGORY	FUTURE REQUIREMENTS		
	RESUPPLY TANKER	OTV	SPACE STATION TANK FARM
<u>ENVIRONMENTAL PHENOMENA</u>			
Long-Term Stratification Effects			X
Soft Outer Shell Performance		X	X
Thermal Coating Degradation		X	X
Micro-Meteroid Protection		X	X
<u>FLUID MANAGEMENT</u>			
Liquid Acquisition Device Refill		X	X
Transfer Line Cooldown	X	X	X
External Tank Scavenging	X		
Receiver Tank Cooldown	X	X	X
Receiver No-Vent Fill		X	X
Refill of Partially Full Tank			X
Propellant Settling		X	
Boiloff Collection			X
Slosh Suppression	X	X	X
<u>ON-ORBIT LOGISTICS</u>			
System Safing	X	X	X
Space Station Interfacing	X	X	X
Space Station Operations	X	X	X
On-Orbit Leak Detection	X	X	X

2.1.2 Define 1985 State-of-the-Art. A technology assessment was then performed for critical technologies. A literature search was conducted to gather relevant data pertaining to each technology. This information was summarized using a standard form documenting the technology assessments. An example of this form, summarizing para-to-ortho H_2 conversion, is depicted in Figure 2-2. All of the technology assessments are provided in Appendix A.

2.1.3 Review Pre-1990 Development Programs. The LTCFSE experiment will evaluate and demonstrate the most advanced technologies possible. Since experiment design and hardware manufacturing will occur in the late 1980's to early 1990's time frame, it is desirable to utilize the technology state from that time frame, rather than current SOA technology when choosing critical technologies for use in the experiment. In order to do this, a review of pre-1990 technology development programs was performed. A list of the development programs surveyed is presented in Table 2-III. A standard form was developed to summarize each of the programs surveyed. An example of this form, summarizing the Passive Orbital Disconnect Strut (PODS) development program is shown in Figure 2-3. All program summaries are provided in Appendix B.

2.1.4 Project 1990 State-of-the-Art Technological advancements from these programs were then utilized to determine the 1990 development level of the technologies. If no development programs were planned for a particular technology, it was assumed that the 1990 SOA was identical to the current level of development.

2.1.5 Select Critical Technologies. Critical technologies to be included in the experiment were chosen from a list of potential technologies. The projected 1990 development level was determined and each technology was ranked according to two criteria: 1. Development Level and 2. Potential Benefit. The development level ranking was based on a scale from one to ten, with a ten being the highest state of development. Table 2-IV shows the scale that was used to perform development level ranking. Potential benefit was ranked on a subjective scale from one to ten, with a one representing a technology that provides little or no benefit and a ten denoting a technology that has a high benefit. The results of this evaluation are presented in Table 2-V for each of the potential technologies.

TITLE: Para to Ortho H_2 Conversion

GENERIC CATEGORY: Thermal Performance

TECHNOLOGY ELEMENTS:
Catalyst bed.

FLIGHT EXPERIENCE:
None.

ADVANTAGES:
Effective use of the endothermic para to ortho conversion increases the cooling capability of hydrogen by approximately 10% as it boils or sublimes and rises to room temperature.

DISADVANTAGES:
Applications of technology not yet developed.

SYSTEM LEVEL DEMONSTRATIONS:
No system level demonstrations of component cooling ability. However, ortho-para converters are used in all H_2 liquefaction plants on a system level.

DEMONSTRATED PERFORMANCE:
Numerous demonstrations of para to ortho conversion have been performed to study effects of flowrate, temperature, pressure and type of catalyst bed. To date, none have provided a demonstration of practical applications, such as cooling a dewar through use of a vapor cooled shield or heat station, or component cooling. Lockheed has performed testing on the effectiveness of a catalyst bed utilizing Apachi-I catalyst. This test measured effectiveness versus flowrate and temperature. Both liquid and solid hydrogen were used as a source of para hydrogen (Reference 2).

DEMONSTRATED RELIABILITY:
The use of a catalyst bed for para to ortho conversion has performed reliably for long-term in hydrogen liquefaction plants. As the same catalyst can be used in para to ortho conversion, its use can be said to be proven reliable over long-term use.

PROBLEM AREAS:
Catalyst contamination.

KEY ISSUES:
Prevention of catalyst contamination, integration on a system level to produce useful cooling.

POSSIBLE IMPROVEMENTS:
Development of system level cooling demonstration.

TECHNOLOGY ASSESSMENT:
Catalyst bed/conversion technology is mature. Technology needs to be developed and matured in terms of practical cooling applications.

RISK ASSESSMENT:
Development towards practical applications would incur minimal risk.

REFERENCES:

1. Sherman, A., Cooling by Para-to-Ortho Hydrogen Conversion, GSC-12770, NASA Tech Briefs, Vol. 7, No. 3, Spring 1983.
2. Nast, T. C. and Hsu, I. C., Development of a Para-Ortho Hydrogen Catalytic Converter for a Solid Hydrogen Cooler, Advances in Cryogenic Engineering, Vol 29, Plenum Press, 1984, pp. 723-731.
3. Clark, R. G., et al, Investigation of the Para-Ortho Shift of Hydrogen, ASD TDR-62-833, prepared by Air Products and Chemicals, Inc., for the Air Force Aero-Propulsion Laboratory.
4. Singleton, A. H., A Rate Model for the Low Temperature Catalytic Ortho-Para Hydrogen Reaction, Doctoral Thesis, Lehigh University, 1968.
5. Singelton, A. H. and Lapin, A., Design of Para-Ortho Hydrogen Catalytic Reactors, Advances in Cryogenic Engineering, Vol. II, Plenum Press, 1966, pp. 617-630.

Figure 2-2. TECHNOLOGY ASSESSMENT FORM.

Table 2-III. PRE-1990 TECHNOLOGY DEVELOPMENT PROGRAMS.

PROGRAM	SPONSORING AGENCY	TIME FRAME
Cryogenic Fluid Mgmt Flight Experiment	NASA-LeRC	1983 - 1993
Zero-G Quantity Gaging	NASA-JSC	1985 - 1987
Oxford Stirling Cycle Cooler	NASA-GSFC	1980 - 1989
Long Duration Exposure Flight Experiment	NASA-LaRC	1984 - 1990
Passive Orbital Disconnect Strut	NASA-ARC	1981 - 1990
Thick Multi-Layer Insulation	AFRPL	1986 - 1989
Long-Term Cryogenic Storage Facility Demonstration Program	NASA-MSFC	1985 - 1986
Cryo Cooler	AFWAL	1965 - present
Multi-Stage Magnetic Refrigerator	AFWAL	1982 - present
Metal Hydride Test Bed	NASA-MSFC	1986 - 1987
Sorption Compressor Refrigeration System	AFWAL	1986 - 1990
Compact Cryogenic Feed System	AFRPL	1986 - 1989

FUTURE CRYOGENIC DEVELOPMENT

PROGRAM TITLE:	Lockheed Passive Orbital Disconnect Strut
CRYOGENIC TECHNOLOGY:	Dual Stage Support
SPONSORING AGENT:	NASA - Ames Research Center
PROGRAM OBJECTIVE:	Development of an elastic deformation disconnect strut to lower on-orbit dewar heat leak.

EXPECTED CRYOGENIC DEVELOPMENT: The current PODS-III design has undergone thermal and structural testing. Lockheed considers the PODS-III system ready for flight applications. They are currently developing a PODS-IV version for application on large tankage systems. PODS-III is currently baselined for use on the Space Infrared Telescope Facility (SIRTF). By the 1990 time frame, PODS should be flight qualified and suitable for application in the long-term storage experiment.

Figure 2-3. TECHNOLOGY DEVELOPMENT PROGRAM SUMMARY FORM.

Table 2-IV. TECHNOLOGY DEVELOPMENT LEVEL SCALE.

CATEGORY	INDEX	DEFINITION
No New Development Required	10	Off the shelf, little or no modification to that which is existing.
	9	Off the shelf design, each item fabricated to individual order and specification.
	8	Known materials, processes, methods and design techniques. No extension to the SOA. Few associated problems.
Extension	7	Materials, processes and methods are presently employed but not to such an extent or magnitude. May be unknown associated problems in design.
	6	Materials, processes or methods have been developed but have not been used in such an application. There are some known problems in design, and some unknown problems may exist.
	5	Apparent solution based upon analysis and physical investigations such as pilot models, simple simulations, etc. Additional development is required to confirm. Many associated problems; many not known.
	4	Apparent theoretical or empirical solution. No actual physical confirmation of the solution. Would require extensive development. Likely many associated problems; few identified.
Beyond SOA	3	Solution looks probable but can only be found with extensive research and development.
	2	There is no reason to doubt a solution can be found if enough time and money are available.
	1	Unknown materials, processes and methods. At this time, there is no indication of a solution to the problem.

Table 2-V. TECHNOLOGY RANKING MATRIX.

TECHNOLOGY	1990 DEVELOPMENT LEVEL[3]	RELATIVE BENEFIT[4]
THERMAL CONTROL TECHNOLOGIES		
Cryogenic Radiators	7	4
Shadow Shields	5	5
Composite Feedlines	7	8[1] 6[2]
Stratification Control	6	7
Cryogenic Heat Pipes	5	9
Dual Stage Support	6	8
Para-Ortho Conversion	6	6
Thick MLI	8	9
Thermodynamic Vent Systems	6	9
Thermal Control Coatings	8	7
Active Refrigeration	5	9
FLUID TRANSFER TECHNOLOGIES		
Mass Flow Meters	6	7
Capillary Acquisition	6	9
Quantity Gaging	7	9
Low Heat Leak Valves	6	7
Low Heat Leak Transfer Lines	8	6
Cryogenic Disconnects	6	6
External Pressurization Loop	6	7
High Pressure Gas Pressurization	7	5
Slosh Suppression	6	5
WEIGHT REDUCTION TECHNOLOGIES		
Soft Outer Shell	7	7
Honeycomb Outer Shell	5	6

1 For supercritical storage
2 For two-phase storage

3 Key:
1 - Least Developed
10 - Most Developed

4 Key:
1 - Least Benefit
10 - Most Benefit

Critical technologies were chosen for inclusion in the experiment based on the ranking performed. Each critical technology satisfied the following criteria:

1. The technology is one that provides obvious benefits in the achievement of long-term storage and transfer of cryogens. This includes basic technologies necessary for the construction of a high performance storage and transfer system, such as thick multi-layer insulation (MLI) and low heat leak transfer lines.

2. The technology will be matured by the 1990's time frame.

3. The technology is required for future orbital cryogenic systems.

In addition, all technologies meeting the above criteria and requiring an on-orbit environment for demonstration were selected. A liquid acquisition device (LAD) is a good example of the technologies in this category. Table 2-VI lists the critical technologies chosen for inclusion in the experiment. A brief description of each technology chosen follows.

Table 2-VI. LTCFSE CRITICAL TECHNOLOGIES.

THERMAL CONTROL TECHNOLOGIES	FLUID TRANSFER TECHNOLOGIES
Stratification Control	Mass Flow Meters
Dual Stage Support	Capillary Acquisition
Para-Ortho Conversion	Low-G Quantity Gaging
Thick Multi-Layer Insulation	Low Heat Leak Valves
Thermodynamic Vent Systems	Low Heat Leak Transfer Lines
Thermal Control Coatings	Cryogenic Disconnects
Active Refrigeration	External Pressurization Loop
WEIGHT REDUCTION TECHNOLOGIES	
Soft Outer Shell	

Stratification Control. Tank fluid stratification results in a higher tank pressure compared to a tank in perfect thermal equilibrium. Fluid mixing that can occur during transfer or station keeping operations will cause sudden pressure drops within the system, complicating control of these systems. Furthermore, evaluation and control of long-term low-g stratification has not been performed. Data from CFMFE and the Shuttle Power Reactant Storage Assembly (PRSA) tanks will provide information only for durations of less than one week in orbit. Evaluation and control of these effects must be understood prior to development of orbital long-term storage systems. The LTCFSE experiment will allow evaluation of long-term stratification effects and the effectiveness of a tank wall heat exchanger for stratification control.

Dual Stage Supports. Conduction heat leak through pressure vessel supports typically constitutes the single largest source of tank conduction heat leak. Dual stage supports meet the requirements of launch and landing loads, but reduce the structural and thermal coupling to the pressure vessel when low-g orbital loads are present. Thus, dual stage supports can greatly enhance dewar thermal performance.

Para-to-Ortho Hydrogen Conversion. Utilizing para-to-ortho conversion in a cryogenic hydrogen storage system can significantly increase the cooling capability of a hydrogen thermodynamic vent system. A great deal of research has been done quantifying this reaction and in determining suitable catalysts for it. The technology is passive, providing a long lifetime, and can be incorporated into a thermodynamic vent system (TVS) with minimal risk.

Thick Multi-Layer Insulation. Thick MLI is the most basic and important technology utilized in the construction of a high performance cryogenic dewar. Multi-Layer Insulation systems exhibit performance levels two orders of magnitude better than other insulations. Therefore, it is the only candidate of interest for long term cryogenic storage applications. Furthermore, it is necessary to demonstrate structural support of these systems during Shuttle launch and landing loads and to evaluate insulation loft performance during extended time periods at low-g.

Thermodynamic Vent Systems. A thermodynamic vent system reduces tank heat leak by removing radiated and conducted heat as it passes through the insulation. Figure 2-4 depicts the improvement in performance obtainable through use of vapor cooled shields. This analysis was generated for the following tank configuration:

- Spherical Tank, Volume = $0.615 m^3$ ($21.7 ft^3$)
- Two-phase Hydrogen, Tank pressure = 101 kPa (14.7 psia)
- S-Glass Strap Support System, A/L = 0.043 cm (0.017 in)
- MLI-14 layers Double Silverized Mylar

A normalized heat leak value of 1.0 is equivalent to 15.3W (52.1 Btu/hr.) In order to achieve heat fluxes low enough for long-term storage without using extremely thick MLI blankets, it is necessary to include such a system. In addition, a vapor cooled shield is needed for integration with para-to-ortho H_2 conversion and active refrigeration testing. Thermodynamic Vent Systems with internal or tank wall heat exchangers can also be utilized to control tank stratification.

Thermal Control Coatings. Dewar outer shell temperature has a significant effect on thermal performance, as shown in Figure 2-4. Therefore, it is desirable to maintain this temperature at a minimum by covering the outer shell of the test tank with a thermal control coating possessing a low solar absorptivity to emissivity ratio α/ϵ . Data from the Long Duration Exposure Facility will aid in the choice of coatings. Long term exposure of the test tank in an orbital environment will also provide data on thermal control coating degradation and its effect on tank thermal performance.

Active Refrigeration. Refrigeration has the potential to completely eliminate boiloff in a cryogenic storage system. This would provide essentially unlimited storage time for such a system. As refrigeration systems require high input power relative to cooling ability, it is still desirable to use a high performance dewar with such a system. The interface between the dewar and refrigerator can be constructed in such a manner that allows for easy interchange of refrigeration systems. This provides a high level of versatility for testing various refrigeration technologies as they mature.

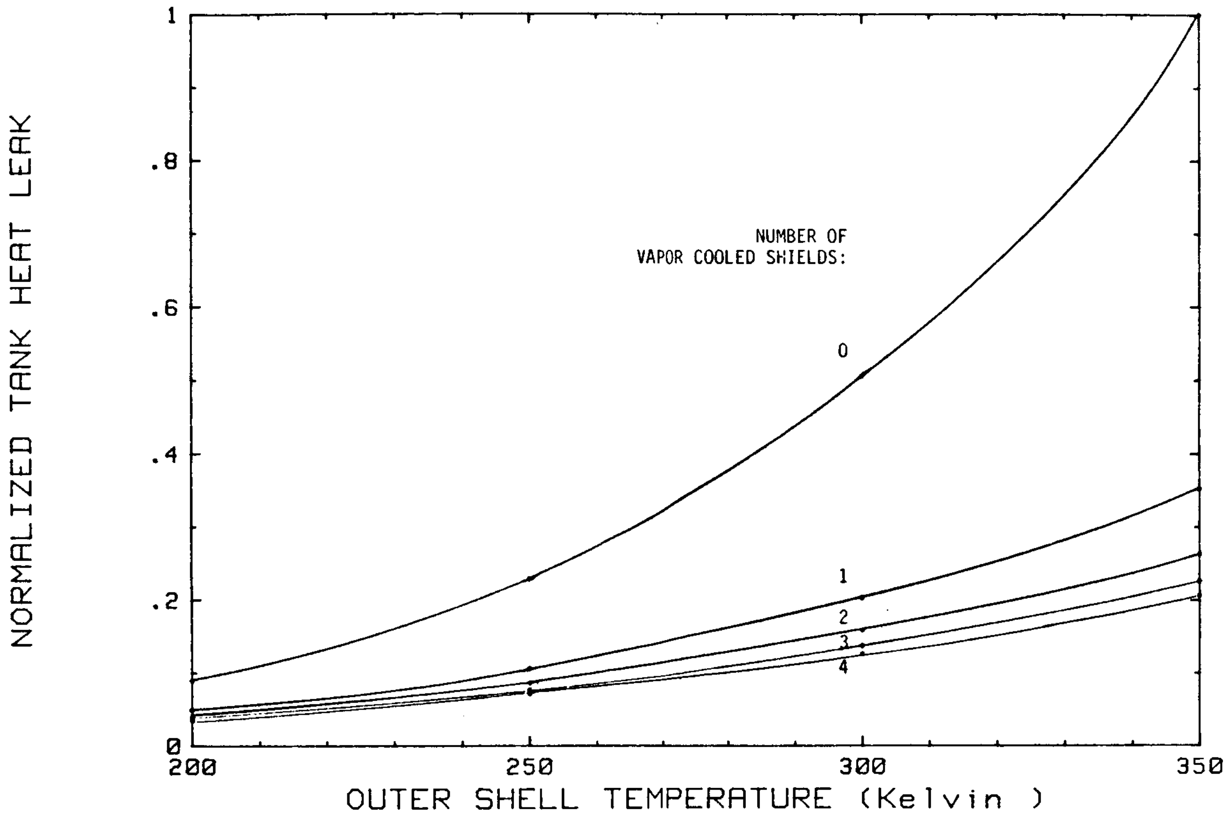


Figure 2-4. H_2 DEWAR PERFORMANCE IMPROVEMENT THROUGH USE OF VAPOR COOLED SHIELDS.

Mass Flow Meters. Mass flowrate is a critical parameter in cooldown of receiver tanks and transfer of cryogens. In addition, integrated mass flowrates can provide a secondary method of calculating mass quantity transferred from a tank. Thus, it is desirable to include mass flow meters in the LTCFSE, as only minimal low-g testing of such systems has been performed.

Capillary Acquisition. A capillary acquisition device will be necessary to effect low-g transfer of liquid from the experimental dewar. It will be necessary to evaluate long-term performance of such a device in a low-g environment and to determine the effect a LAD has on the long-term storage of cryogens.

Low-G Quantity Gaging. Low-g quantity gaging is a technology that is important in applications of long-term cryogenic storage and transfer. It is required for gaging boil-off and fluid transfer operations. There will not have been a long-term test of a low-g quantity gaging system by the time the experiment is deployed. Therefore, inclusion of a quantity gaging system is highly desirable both to perform long-term testing of the system itself, and to evaluate the effects of a quantity gaging system on long-term storage dewar performance.

Low-Heat Leak Transfer Lines. At the current flight cost of $6600/kg to transfer materials to the Space Station orbit, conservation of cryogen produces a high economic benefit. Utilization of low heat leak valves and transfer lines will minimize cryogen losses during fluid transfer. In addition, decreasing boiloff during fluid transfer has additional benefits, such as decreased line pressure drop, which in turn decreases pressurant requirements and overall system mass.

Cryogenic Disconnects. Refueling of orbital systems will require fluid disconnects that attach to the servicing depot. These disconnects must be low heat leak and must have minimal to zero leakage when disconnected. Disconnects will be required on the LTCFSE experiment not only to demonstrate the technology, but also to achieve modularity in experiment design.

External Pressurization Loop. An external pressurization loop is a system that provides pressurant for cryogen expulsion by utilizing conditioned cryogen from the parent system. This avoids the need for high pressure gas supply vessels. Such a system is highly desirable for a permanent space-based supply depot, and thus will be demonstrated on the LTCFSE experiment.

Soft Outer Shell. Supply depot tanks will typically be very large, on the order of 100 m^3 (3530 ft^3) or more. The Tethered Orbital Refueling Facility (TORF) study performed by Martin Marietta (Reference 2) baselines a 139 m^3 (4900 ft^3) H_2 tank, holding a total mass of approximately 8620 kg (19,000 lbm) of cryogen. The corresponding O_2 tank has a volume of 36.5 m^3 (1290 ft^3), holding 36,774 kg (81,000 lbm) of O_2. The total dry weight of the H_2/O_2 tankset is 12,860 kg (28,350 lbm). Such tanks can not be launched by the Shuttle loaded with cryogen due to Shuttle payload mass constraints. As a result, soft outer shell tanks may be utilized without the thermal performance penalty incurred from an integrated MLI purge system. Demonstration of a soft outer shell tank that will be launched empty can be performed using the LTCFSE receiver tank. In addition, thermal performance and micrometeroid protection systems can be evaluated and compared to the hard outer shell system that is baselined for the LTCFSE supply tank.

Honeycomb outer shell technology was an alternate weight reduction technology investigated during Task I. It was not chosen as critical technology due to the absence of future development programs and because this technology may be developed and demonstrated without orbital testing. However, preliminary investigations (Ref. 1) indicate honeycomb outer shells may be easier and less expensive to manufacture than a hard outer shell. If this technology is further developed, it would be an attractive alternative for use on the LTCFSE supply tank.

2.1.6 Define Technology Development Experiments. The critical technologies that were identifed for inclusion in the LTCFSE experiment were grouped into compatible technology sets to be demonstrated in different phases of the experiment. Two basic criteria were utilized to group the technologies:

1. The technologies are operationally compatible.
2. The technologies are at similar stages of development and will be mature enough for inclusion in the experiment.

Using these criteria, the LTCFSE experiment was divided into three phases as follows:

Phase I - Passive Thermal Technologies
Phase II - Fluid Transfer Technologies
Phase III - Active Refrigeration Technologies

Table 2-VII shows the technologies that will be demonstrated in each phase of the experiment. A zero in a column indicates that although this particular technology was demonstrated in a previous phase, additional hardware will be added that will further demonstrate the same technology. For example, a TVS will be utilized in the Phase I supply tank. However, the hardware added during Phase II will contain a receiver tank which also has a TVS that provides further demonstration of this technology.

Table 2-VII. TECHNOLOGY DEMONSTRATION MATRIX.

TECHNOLOGY	PHASE I PASSIVE THERMAL	PHASE II FLUID STORAGE & TRANSFER	PHASE III ACTIVE REFRIGERATION
Thermal Control:			
1. Stratification Control	X	O	
2. Dual Stage Support	X		
3. Para-Ortho Conversion	X		
4. Thick MLI	X	O	
5. Thermodynamic Vent System	X	O	
6. Thermal Control Coatings	X	O	
7. Active Refrigeration			X
Fluid Transfer:			
8. Mass Flow Meters	X	O	
9. Capillary Acquisition	X	O	
10. Low-G Quantity Gaging	X	O	
11. Low Heat Leak Valves		X	
12. Low Heat Leak Transfer Lines		X	O
13. Cryogenic Disconnects		X	O
14. External Pressurization Loop		X	
15. Slosh Suppression	X	X	
Weight Reduction:			
16. Soft Outer Shell		X	

O Additional enhancement of technology acquired during this phase

2.1.7 Preliminary Scaling Analysis. A preliminary scaling analysis was performed to determine an approximate size that will be adequate to demonstrate long-term cryogenic storage. Section 2.2 scrutinizes this analysis more closely with items such as experiment length and cryogen requirements taken into consideration. The primary purpose of this analysis was to approximate experiment size for input to Task II. This in turn allowed the experiment requirements to be documented and put into the NASA Space Station data base in a timely fashion.

The most critical parameter involved with long-term cryogenic storage is heat leak, or more specifically, boiloff. In terms of boiloff, percent boiloff per unit time determines tank storage lifetime. Percent boiloff per unit time can be expressed in heat leak terms by heat leak per unit volume (Q/V). Both percent boiloff and Q/V are functions of tank volume. This is primarily due to the tank surface area to volume ratio, A/V and heat conduction from supports, both of which are volume dependent.

A parametric heat leak analysis to determine Q/V as a function of tank volume was performed using Beech Aircraft's cryogenic tank analysis program Liquid Cryogen Tank. The analysis was performed for a high performance dewar configuration with volumes ranging from 0.14 m^3 (5 ft^3) to 142 m^3 (5000 ft^3). The dewar configuration is one that will be similar in design to the LTCFSE Phase I tank. That is, it is a high performance dewar, but does not use technologies such as active refrigeration to enhance dewar performance. The basic system parameters utilized in the analysis are as follows:

1. Two vapor cooled shields
2. MLI: Double silverized mylar/silk net
   Emissivity = 0.022, 20 layers/inch
3. Dual stage supports utilized
4. Tank fluid is two-phase hydrogen at 345 kPa (50 psia)
5. Outer shell temperature is 300 K (540°R)

The results of this analysis are presented in Figure 2-5 as a plot of normalized Q/V versus tank volume. A normalized value of 1.0 corresponds to a Q/V of 0.10 W/m^3 (0.01 Btu/hr-ft^3). A tank volume of 139 m^3 (4900 ft^3) was chosen as the basis for the scaling analysis. This is the volume of a hydrogen tank required to refuel two OTV tanks, and is the baseline H_2 tank volume used in the TORF study performed by Martin Marietta (Ref 2). Since it is impractical to use such a large tank for the Long-Term Storage Experiment, one that can be scaled 2:1 on a heat leak basis was chosen. As seen in Figure 2-5, the normalized Q/V for a 139 m^3 (4900 ft^3) tank is 0.57. A tank with twice this heat leak has a normalized Q/V of 1.14. Such a tank, shown in Figure 2-5, has a volume of 11.3 m^3 (400 ft^3). This tank is much more manageable in terms of size, fits easily in the shuttle cargo bay, and will have significantly lower flight costs. Thus, data from all phases of the experiment will be directly scalable to the refueling facility tanks.

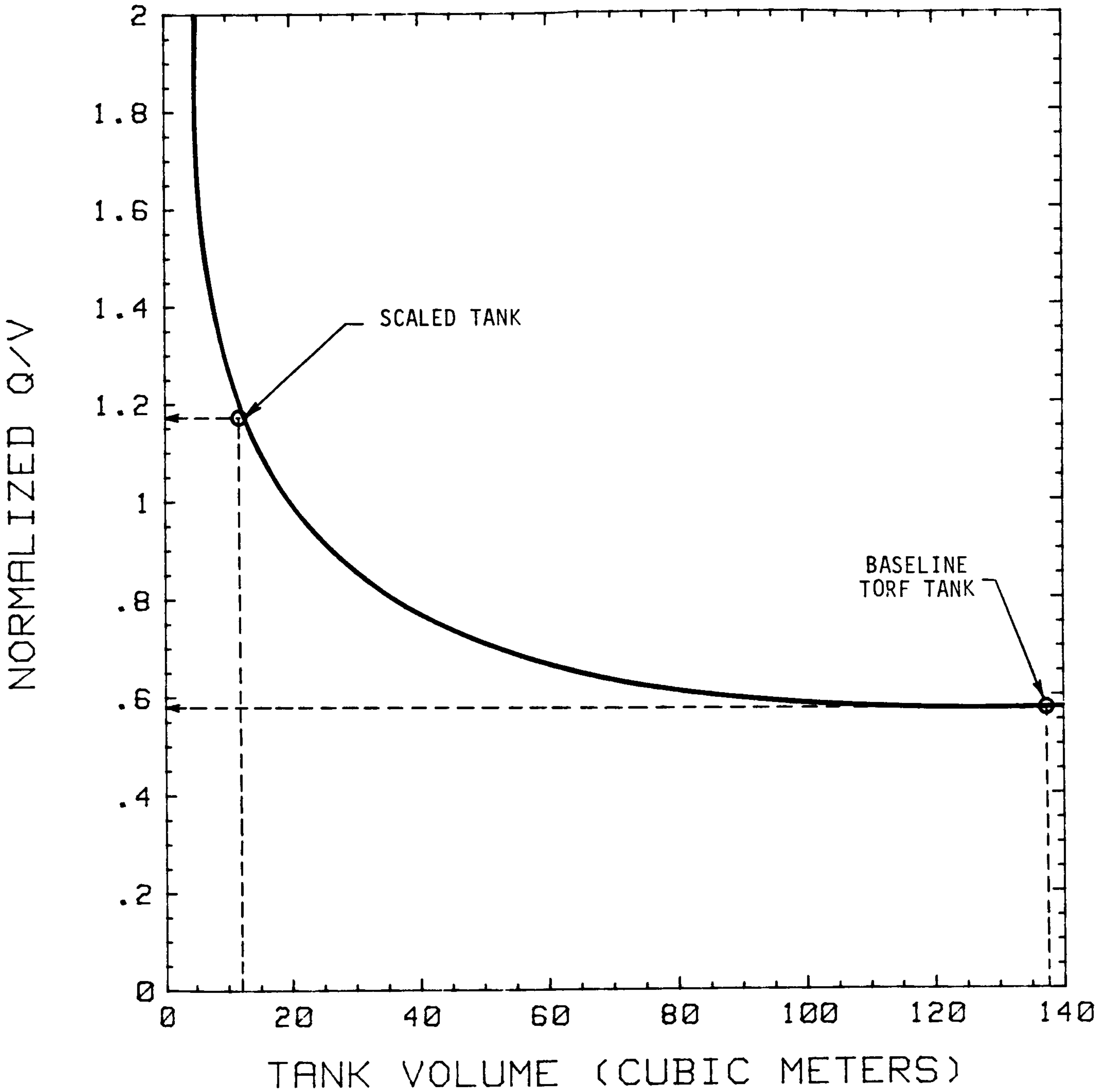


Figure 2-5. NORMALIZED Q/V VERSUS TANK VOLUME.

2.1.8 Time-Phased Test Plan. A time-phased test plan was prepared that sequenced the three phases of the experiment and scheduled significant activities, such as experiment deployment and reconfigurations. The maturity of each group of technologies was the primary consideration used to sequence the phases. Technology groups that were most mature were scheduled to be tested first. The test plan for the three experiment phases is presented in Figure 2-6. Initial deployment is scheduled for 1993, which is the scheduled date to begin operation of the IOC Space Station. Deployment, reconfiguration hardware checkout and length of experiment operations are shown for each phase. Experiment retrieval is not depicted, as it may be desirable to maintain the experiment on Space Station, either for further experimentation or for reuse on Space Station as a cryogenic supply system. Phase II testing will contain numerous transfer operations, and receiver tank Beginning-of-Life (BOL) and End-of-Life (EOL) thermal testing. These Phase II operations are presented in detail on Figure 2-7.

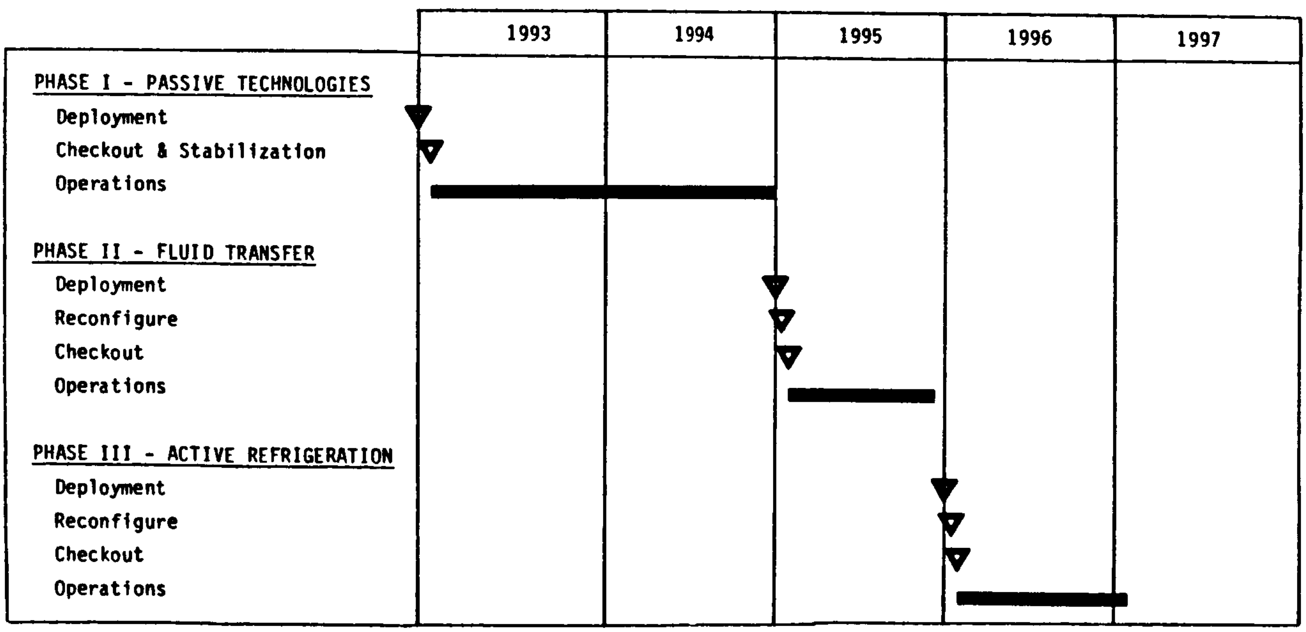


Figure 2-6. EXPERIMENT TEST PLAN.

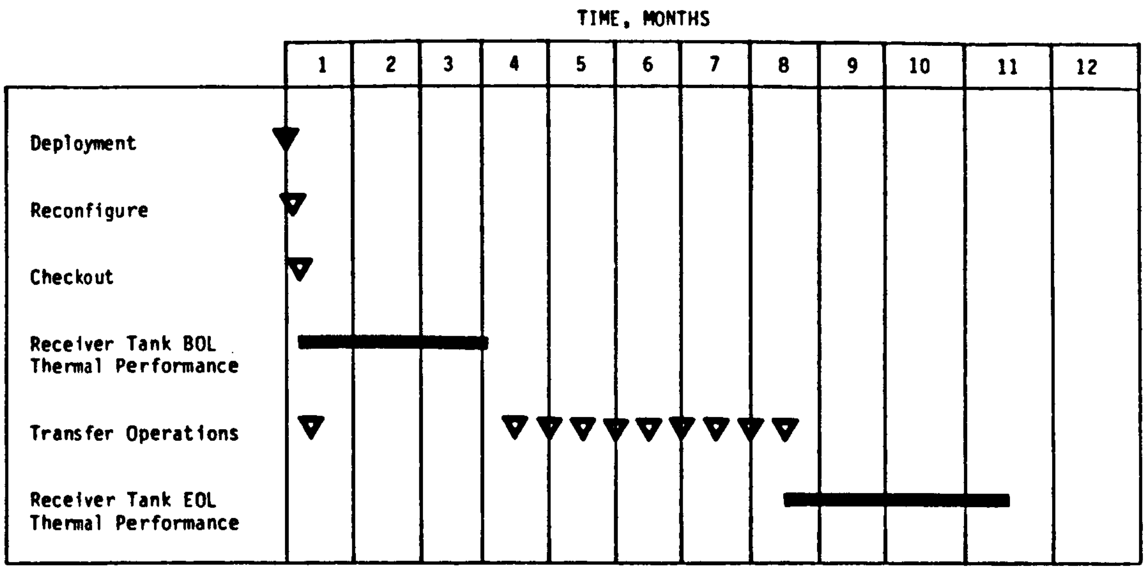


Figure 2-7. PHASE II DETAILED TEST PLAN.

2.2 Task II - Determination of Experimental Requirement. The objective of Task II was to provide a preliminary design for the LTCFSE experiment. This will allow documentation of experimental requirements to be entered into the Space Station data base as soon as possible. The preliminary design was based on inputs from Task I and from the interface restrictions summary and available hardware review that were performed early in the Task II effort. The preliminary design performed in this task was updated in Task IV. Details of the experiment design from Task IV will be presented in Section 2.4.

2.2.1 Interface Restrictions Summary. Before beginning the Phase II design effort, restrictions due to interfaces required during STS prelaunch, launch, deployment and recovery and Space Station deployment and operations were investigated to identify potential design restrictions. The following areas that contain potential interface restrictions were investigated and include evaluation of any impact on the experiment design:

Size. The Shuttle cargo bay provides the most limiting restrictions - 2.34m (92 in) radius, 19.8m (65 ft) long. This will impose no constraints on the current experiment design. Support structure for the experiment is designed to fit the standard Shuttle trunnion pin mounting fixture.

Mass. Shuttle lift capacity restricts Space Station launch mass to no more than 17,234 kg (38,000 lbm) (Reference 3). This imposes no constraints on the experiment design, which has a total mass of approximately 3200 kg (7,000 lbm). This total mass is for all three experiment phases. The largest single mass that will be launched is Phase I, with a total mass of approximately 1900 kg (4,300 lbm). Shuttle center of gravity constraints (Reference 3) indicate a center of gravity location of no more than STS Station Number 1188 for the Phase I package.

Power. The highest power requirement for the LTCFSE experiment is 2.5 kW, during Phase III. This is 5% of the 50 kw allotted for IOC Space Station users. Rocketdyne Space Station design personnel working on the Phase B Work Package IV have indicated 2.5kW of power can most likely be made available to one user.

Data Acquisition. The Data Management System (DMS) currently baselined for Space Station provides adequate capability for use with the LTCFSE experiment. Experiment sensors and any signal conditioning units required will be designed or purchased to be compatible with the Space Station DMS.

Acceleration. Worse case acceleration environments occur during launch in the Shuttle cargo bay and are listed in Table 2-VIII. The experiment will be designed for these loads, with no impact on experimental capability.

Table 2-VIII. ANTICIPATED SHUTTLE LAUNCH LOADS (Reference 3).

Steady State Acceleration	
Flight (Ascent/Descent)	3.2 g (limit)
Lift Off/Landing	6.0 g (limit)
Emergency Landing	4.5 g (ultimate)
Vibration Environment	
Random:	
Root Mean Square G-level	8.72
Power Spectral Density Peak (g2/Hz)	0.15
Duration (sec)	190 each axis
Sinusoidal:	
Swept Sine	5 - 35 Hz $\pm$ 0.25g (peak)

Pointing. There are no pointing restraints for the experiment.

Teleoperation. Teleoperation utilizing the Shuttle and Space Station Remote Manipulator Systems (RMS) will be utilized for experiment deployment and retrieval. The only constraint this implied in design was the addition of RMS grapple fixtures to the experiment structure.

Extravehicular Activity. Extravehicular Activity (EVA) will be required during experiment deployment and retrieval to connect and remove experiment interfaces. Maximum EVA time is currently 8 hours. The only constraint this places is the need for multiple EVAs to complete some activities. Because of the cost ($200,000/hr) and the high demand for EVA hours, requirements are minimized in the design.

Servicing. The experiment will be designed for minimal servicing. Currently, no servicing is required for normal experiment operation.

Environment. The LTCFSE experiment must be designed to survive a variety of environments. These include ground handling and servicing, Shuttle payload bay ground, launch, and on-orbit environments and finally, the Space Station orbital environment. Thermal and structural analysis must be conducted on the experiment design to ensure proper experiment survival and operation in these environments. Micrometeroid and atomic oxygen effects must also be taken into account in experiment design. The hard outer shell and MLI in the Phase I supply tank will provide adequate micrometeroid protection against pressure vessel (PV) rupture. The Phase II receiver tank will contain a micrometeroid shield, which in conjunction with the MLI will also provide adequate micrometeroid protection against PV rupture (Reference 4). The micrometeroid shield will also provide protection for the MLI against the atomic oxygen environment, which is known to cause rapid degradation of MLI. Thermal control coatings which are to be used on the supply and receiver tanks to lower tank heat leak must be resistant to atomic oxygen degradation. A silverized teflon sandwich coating, consisting of a layer of silver sandwiched between two teflon layers, has shown high resistance to degradation and is the prime candidate for this application.

2.2.2 Available Hardware Review. A review of existing hardware that can have potential use on the LTCFSE experiment was performed. Use of existing hardware and/or designs will reduce program cost and schedule length. Table 2-IX lists the hardware items that were assessed for use in the experiment. Each assessment was summarized on a standard form, an example of which is shown in Figure 2-8. For each item, a general description was provided, along with specifications that are pertinent to its potential application. The availability of the hardware was also defined. Finally, the advantages and disadvantages of using the hardware were summarized, along with a recommendation as to whether it should be incorporated into the experiment. A compilation of the available hardware reviews performed is presented in Appendix C. The results from this hardware review was used in Task IV in order to generate a more detailed conceptual design.

Table 2-IX. AVAILABLE HARDWARE REVIEWED.

Oxygen Thermal Test Article (OTTA)
Hydrogen Thermal Test Article (HTTA)
Cryogenic Fluid Management Flight Experiment (CFMFE) Receiver Tank
Cryogenic Fluid Management Flight Experiment (CFMFE) Supply Tank
Power Reactant Supply Assembly (PRSA) Hydrogen Tank
Earth Limb Measurement Satellite (ELMS) Tank
Fuel Cell Servicing System (FCSS)
Centaur GSE Loading System
Centaur Orbiter Modification Kit

2.2.3 Preliminary Conceptual Design. A preliminary conceptual design of the LTCFSE experiment was completed in Task II. The primary purpose of this design was to allow preliminary experiment requirements to be entered into the Space Station data base at an early date. The design and requirements were subsequently updated during the Task IV detailed design effort. A brief summary of the Task II experiment design is presented in Table 2-X. Isometric views of the preliminary design produced during Task II are shown in Figure 2-9. A detailed description of the final design will be presented in Section 2.4.

Table 2-X. PRELIMINARY DESIGN SUMMARY.

Supply Tank - 11.3m3 (400 ft3) Cylindrical Tank, Hemispherical Heads
Receiver Tank - .38m3 (13.4 ft3) Cylindrical Tank, Hemispherical Heads
Pressurization System - External pressurization loop with H_2/O_2 gas generator and heat exchanger for pressurant gas conditioning mass - 181 kg (400 lbm)
Active Refrigeration Unit - 5 W cooling at 20 K
Total System Mass - 3450 kg (7600 lbm)

AVAILABLE HARDWARE REVIEW

Hardware: Power Reactant Supply Assembly (PRSA) Hydrogen Tank.

Availability: The PRSA tanks are currently being flown on the Shuttle. They are currently not available for use.

Description: The PRSA hydrogen tank is a .615 m^3 (21.7 ft^3) flight qualified hydrogen dewar. See Figure C-6 for more details.

Potential Application: Use as a receiver tank in Phase II fluid transfer experiments.

Critical Specifications:

- Volume - .615 m^3 (21.7 ft^3)
- Design Pressure 1.97 MPa (285 psia)
- Spherical Dewar - 1.2 m (47.24 in.) O.D.
- One Vapor Cooled Shield
- Strap Support System
- 14 Layers double silverized mylar MLI
- Heat Leak - 2.6 watts (8.8 BTU/hr)
- Wet Weight - 146 kg (322 lbm)

Advantages

- Use of available design will reduce experiment cost and development time.
- Tank has been Shuttle flight qualified.

Disadvantages

- Large amount of rebuild needed to reconfigure as a receiver tank, particularly internal to the pressure vessel.
- Pressure vessel mass to volume ratio (m/V) is higher than desired.

Recommendation (including required modifications):

The following modifications would be required to reconfigure as a suitable receiver tank:

- Addition of a thermodynamic vent system external heat exchanger and Joule-Thomson valve.
- Addition of more MLI
- Addition of spray nozzle system internal to the pressure vessel.
- Addition of more instrumentation.
- Addition of Liquid Acquisition Device

Since the hardware is not available, and PV m/v ratio is higher than desired, utilization of the PRSA H2 tank is not recommended.

Figure 2-8. AVAILABLE HARDWARE REVIEW FORM.

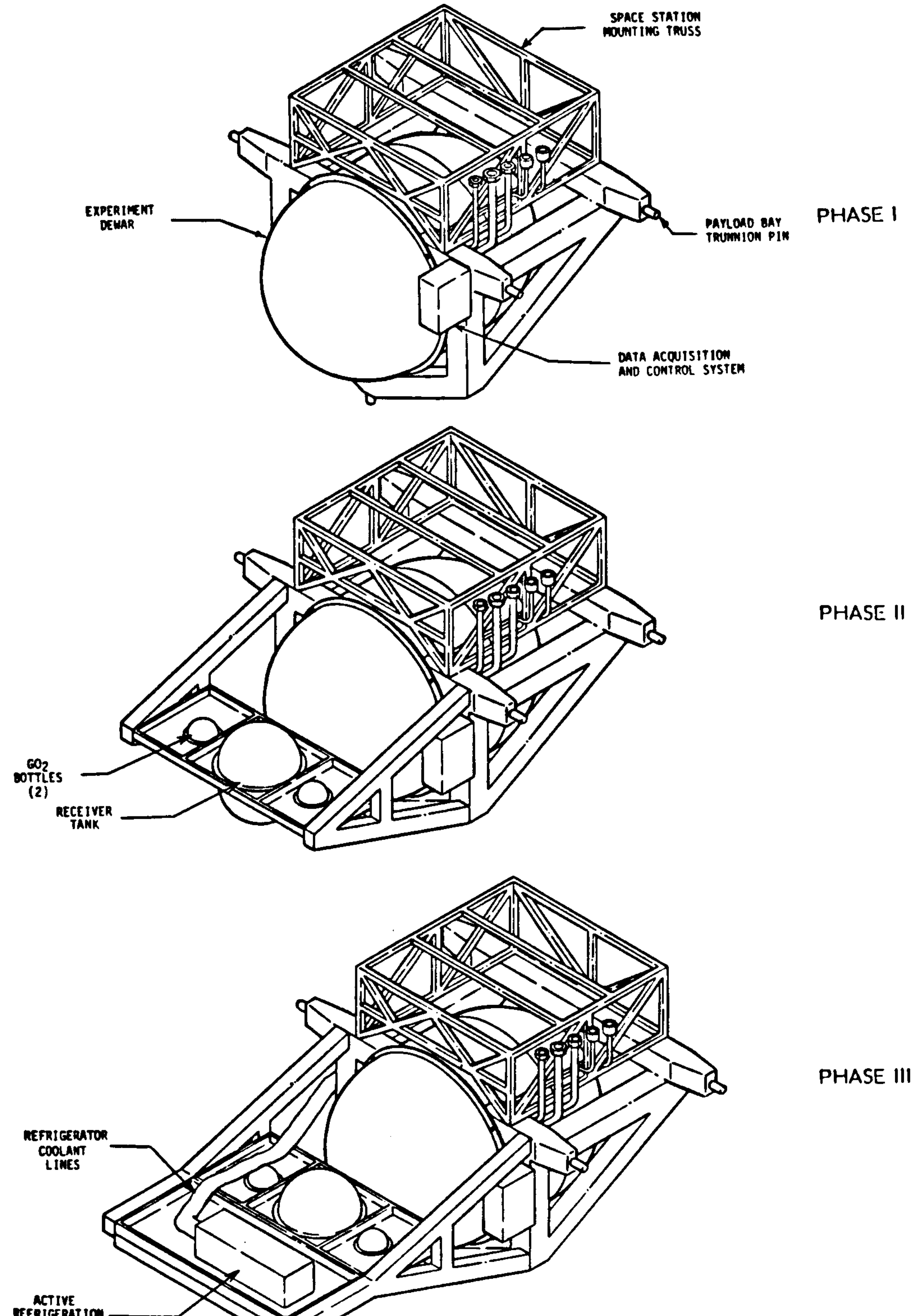


Figure 2-9. LTCFSE PRELIMINARY DESIGN CONCEPTS.

2.3 Task III - Documentation of Experiment Requirements. The objective of Task III was to document the LTCFSE experiment requirements. These requirements were then entered into the Space Station data base. Documentation was performed using TDAG and MRDB forms. These forms were initially completed utilizing the Task II preliminary design requirements. The forms were updated at a later date based on the revised Task IV detailed design. The updated TDAG and MRDB forms are presented in Appendices D and E, respectively.

2.4 Task IV - Detailed Conceptual Equipment Design. The objective of Task IV was to develop a detailed conceptual design of the experiment hardware. This design further developed and refined the preliminary Task II design. During this effort, cryogen requirements were reviewed in detail. Specific items reviewed include receiver tank and transfer line cooldown requirements and total tank boiloff requirements. In addition, safety, control, interface and contamination issues were reviewed for any potential impact on system design. Detailed sketches and specifications were produced for all major components of the systems. An experiment location on Space Station for the hardware, along with a suitable method for attachment to the Space Station, was determined. Upon completion of this detailed conceptual design, the TDAG and MRDB forms were updated to reflect changes from the preliminary Task II design.

2.4.1 Develop Configuration. The first step in producing the detailed design was to develop the overall experiment configuration. Experiment objectives and requirements were reviewed for all three program phases to ensure that hardware designed for earlier phases will meet all requirements for the later phases. The approach utilized to develop the experiment configuration is depicted in Figure 2-10. The receiver and supply tanks were sized based on cryogen requirements, scaling considerations, and experiment objectives. Use of available hardware was also a primary consideration in this sizing effort. A trade study was performed to select a pressurization system for use in the Phase II fluid transfer operations. Finally, refrigerator interface requirements for Phase III were considered to ensure Phase I and III compatibility. Once these basic parameters were defined, detailed design of the experiment was performed.

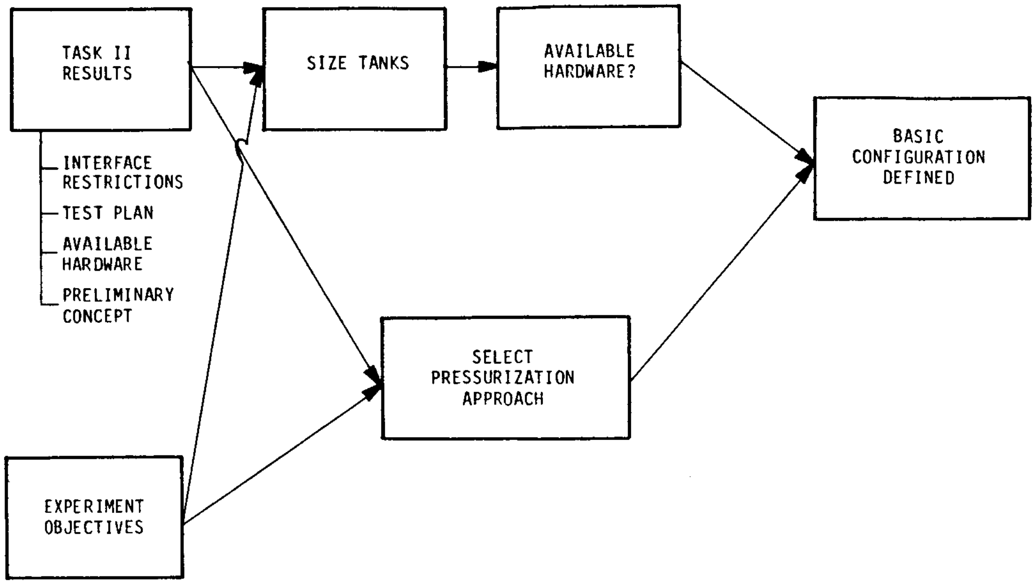


Figure 2-10. TASK IV - CONFIGURATION DEVELOPMENT APPROACH.

The Phase I supply dewar was sized to provide cryogen without resupply for all three phases of the experiment. This decision was based on the following reasons: 1) one or more resupply missions would increase experiment cost and complexity and 2) use of a larger supply dewar with adequate cryogen for all phases of the experiment will more closely approximate the thermal performance of the large supply tanks proposed for the Space Station tank farm. Subcritical (two phase) hydrogen was chosen as the test cryogen since LH_2 will be required for OTV refueling missions, and use of H_2 allows demonstration of para-to-ortho H_2 conversion. In addition, H_2 has a much lower density than O_2, reducing experiment launch costs, and is a safer cryogen to use than O_2.

2.4.1.1 Receiver Tank Selection. The volume of the Phase I supply dewar is based primarily on two factors: 1) supply tank boiloff over the total experiment duration, and 2) receiver tank and transfer line cooldown requirements and receiver tank boiloff during Phase II operations. Thus, receiver tank volume must first be determined prior to sizing the supply tank. A primary parameter of interest in receiver tank sizing is the ratio of pressure vessel mass to volume. This ratio determines the amount of cryogen per receiver tank unit volume required to perform cooldown of the receiver tank prior to fill operations. In normal gravity tank fill operations, tank cooldown is accomplished during the tank fill. Cryogen boils off during fill, cooling the tank wall, and the resulting vapor is simply vented. However, in low-g fill operations, there is no effective, simple method for ensuring that only vapor will be vented during a tank fill. Thus, a procedure known as a no-vent fill must be performed.

This fill can consist of numerous pre-chill cycles to cool the tank wall, followed by a tank chill and fill operation. Tank pre-chill is accomplished by injecting a small amount of cryogen into the receiver tank with the vent line closed. The injected cryogen boils and becomes superheated, cooling the tank wall and increasing the tank pressure. After all the cryogen has evaporated and the cryogen temperature approaches that of the tank wall or the tank maximum pressure is reached, the tank is vented, and another pre-chill cycle begins. After tank pre-chill has been completed, the receiver tank wall temperature will be slightly higher than the desired saturated liquid temperature. At this point, after the final vent cycle, tank chill and fill occurs without venting. Final tank wall cooldown to the desired saturation liquid temperature occurs during tank fill. This type of pre-chill cycle also minimizes cryogen mass required for cooldown, because a significant amount of cooling is achieved via the sensible heat of the vapor as it becomes superheated.

Large Space Station-based refueling tanks will have m/V ratios of approximately 6 kg/m^3 (0.375 lbm/ft^3). The receiver tanks baselined for use on the CFMFE have m/V ratios of 80 kg/m^3 (5.0 lbm/ft^3). It was desired to utilize a receiver tank for the LTCFSE that satisfies the following requirements:

1. The tank should not have critical parameters such as m/V that are a duplicate of CFMFE hardware. This will ensure CFMFE test data are not needlessly duplicated.

2. The receiver tank m/V should be as close to the OTV projected m/V (approximately 6 kg/m^3) as possible, yet still be manageable in size.

3. Available hardware or designs should be used if possible.

4. Geometric similitude should be generally observed.

Figure 2-11 shows a plot of tank m/V variations with volume for tank volumes up to 100 m^3 (3530 ft^3). The lower line, for spherical tanks, represents the theoretical minimum m/V value possible for a tank. PV wall mass is proportional to PV surface area and a sphere has the smallest surface area per unit volume (A/V) of any enclosure. Thus, for a given minimum wall thickness, a spherical tank will have the smallest possible m/V. The upper lines represent theoretical m/V ratios for cylindrical tanks having length over diameter ratios (L/D) of 2.0 and 4.0. These curves were generated using the Beech Aircraft Conventional Tank Program (Reference 5), that performs tank sizing computations. These m/V curves were based on the following receiver tank parameters:

Design Pressure - 345 kPa (50 psia)
Minimum Wall Thickness - .89 mm (.035 in)
Ultimate Strength Factor of Safety - 2.0
Yield Strength Factor of Safety - 1.5
Pressure Vessel Material - AL 2219

A design pressure of 345 kPa (50 psia) was chosen to allow for pressure rises during the pre-chill cycles. Comparing the theoretical minimum lines to available hardware data, it can be seen the HTTA and ELMS tanks have reasonable m/V values. However, the HTTA has a volume of 22.8 m^3 (806 ft^3), which is much larger than the anticipated experiment receiver tank volume.

Based on this plot, and the receiver tank requirements listed above, the ELMS tank design was chosen for use as the Phase II receiver tank. This tank has a reasonable volume, 1.27 m^3 (45 ft^3) and a relatively low m/V of 34.4 kg/m^3 (2.1 lbm/ft^3).

2.4.1.2 Tank Cooldown Losses. Once the receiver tank was chosen, the cryogen requirements needed for tank cooldown prior to fill were calculated. The Beech Aircraft Tank Cooldown Analysis Program (TNKCAP), Reference 6, which models the no-vent fill process, was utilized to predict this requirement. TNKCAP uses a three node receiver tank model as shown in Figure 2-12. The analysis of the charge-hold-vent pre-chill process begins with user specified tank inlet conditions, upstream of the cooldown spray nozzles. This inlet fluid is then isenthalpically expanded through the nozzles to tank pressure.

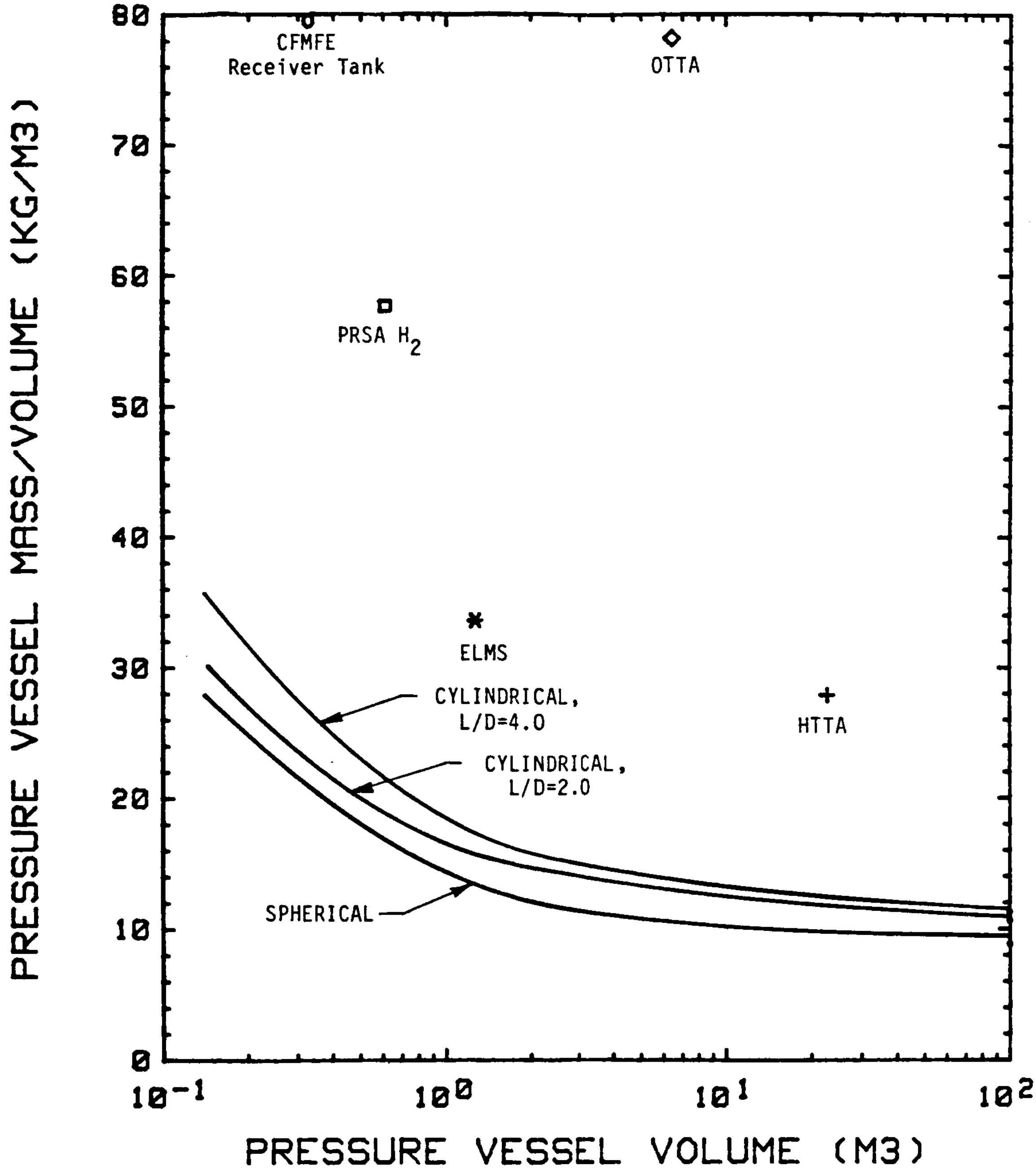
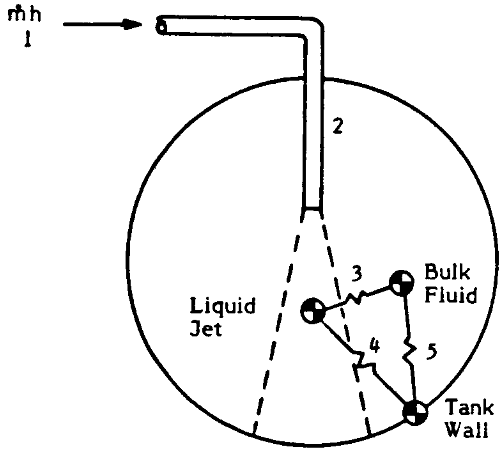


Figure 2-11. RECEIVER TANK m/V VERSUS VOLUME.



THERMODYNAMIC PROCESS

1. **Mass Flow Into Tank.**
2. **Isenthalpic Expansion.**
3. **Convective Heat Transfer From Liquid Drops to Bul Fluid.**
4. **Heat Transfer Between Liquid Jet and Tank Wall:**
   a. **Nucleate Boiling**
   b. **Film Boiling**
5. **Heat Transfer Between Bulk Fluid and Tank Wall.**

Figure 2-12. RECEIVER TANK THERMODYNAMIC MODEL.

The heat transfer rate between the bulk fluid and the jet node is calculated and the amount of liquid vaporized is determined. The droplet size at the tank wall is computed and the heat transfer rate between the wall and jet is calculated as well as the mass of liquid vaporized. The mass of inlet fluid and its average enthalpy is then added to the fluid node. The heat transfer rate between the wall and fluid node is calculated based on forced convection. Energy and mass balances are performed on the tank wall and fluid nodes. Using the tank wall energy balance and a user input table of integrated tank wall specific heat, the new tank temperature is determined.

From the average tank density and pressure, the new cryogen temperature is determined. The temperature calculation assumes that bulk liquid and vapor are in thermal equilibrium. If the tank fluid node density indicates that the liquid is collecting in the tank, the mass fraction of liquid is computed.

This process is repeated until the tank wall and fluid node temperatures are within 10% of each other, or a user specified maximum tank pressure has been reached. When the tank wall temperature reaches a specific target temperature, the no-vent tank chill and fill begins.

Graphical results of the TNKCAP simulation are presented in Figures 2-13 through 2-18. Figures 2-13 through 2-15 show an expanded view of the pre-chill portion of the no-vent fill and the beginning of the tank fill process. The pre-chill portion of the cycle lasts approximately 300 seconds, at which point tank chill and fill begins.

Figure 2-13 depicts cryogen and PV wall temperature variation with time. The cryogen temperature alternately rises and falls as the cryogen evaporates and superheats and is then vented. Simultaneously, the PV wall temperature gradually drops as it is cooled by the cryogen. This plot shows that tank vent occurs when the cryogen temperature is within 10% of the PV wall temperature. At 300 seconds, the tank wall reaches 70K (the target temperature), and the tank chill process begins followed by tank fill.

Figure 2-14 shows tank pressure variation with time for pre-chill. As the cryogen evaporates and rises in temperature, the tank pressure also rises. During the vent portion of the cycle, the tank is vented down to 13.8 kPa (2 psia) and then recharged.

Varying parameters such as final vent pressure will change the effectiveness of the chilldown cycle. It is these types of parameters that will be changed for each Phase II tank fill operation to determine the most effective tank fill procedure.

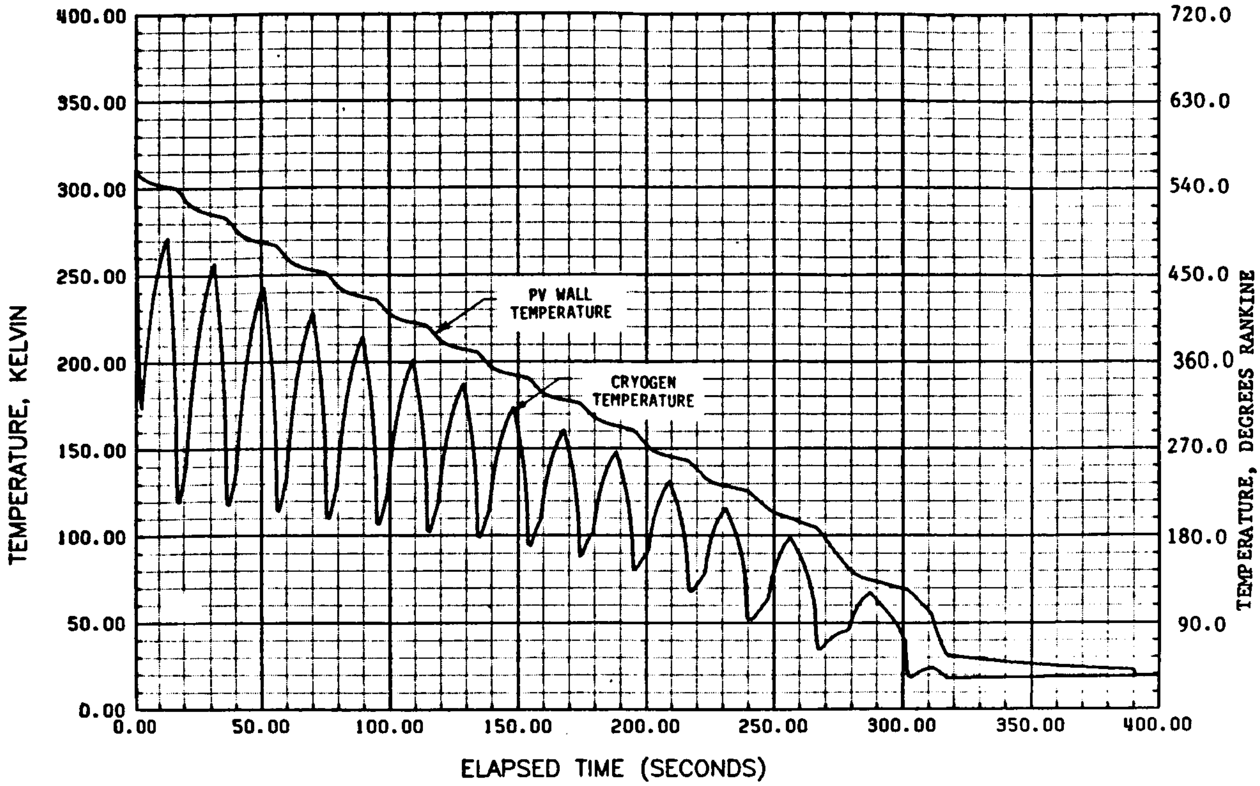


Figure 2-13. TEMPERATURE VS. TIME - LTCFSE RECEIVER TANK CHILLDOWN.

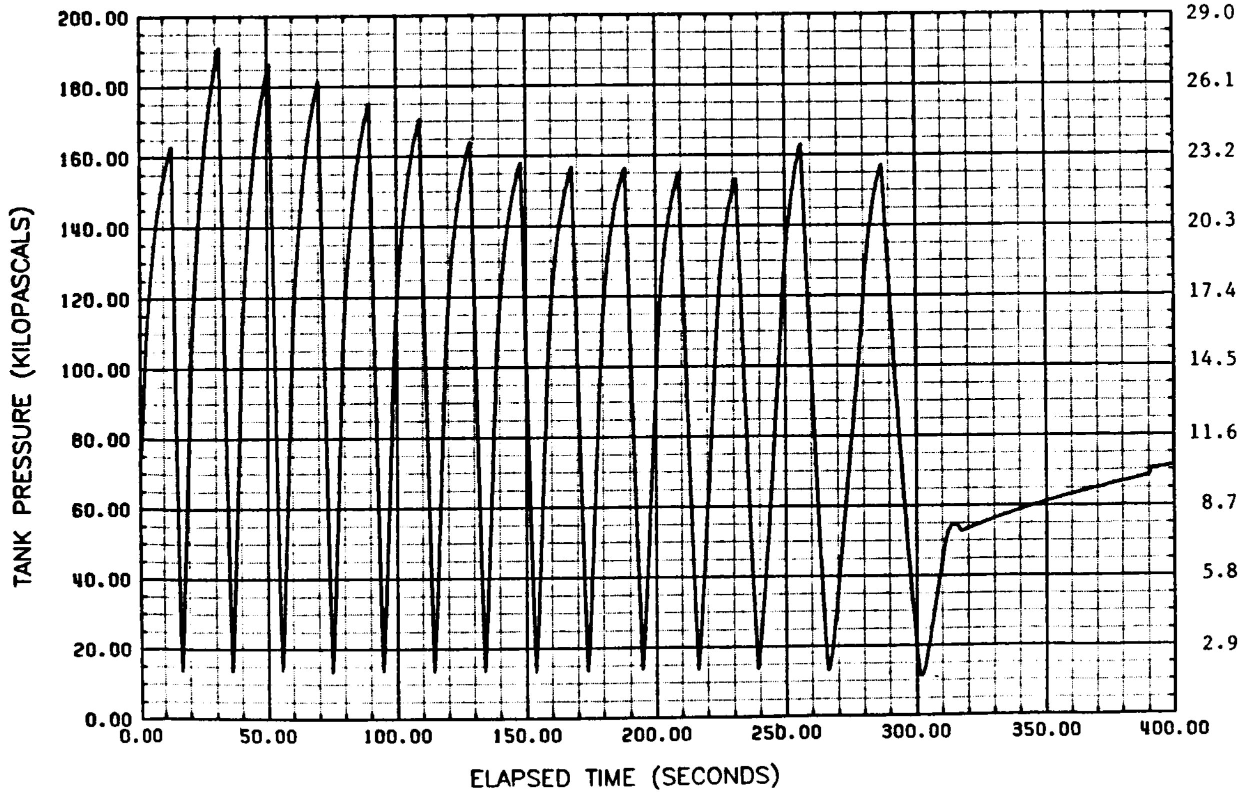


Figure 2-14. TANK PRESSURE VS. TIME - LTCFSE RECEIVER TANK CHILLDOWN.

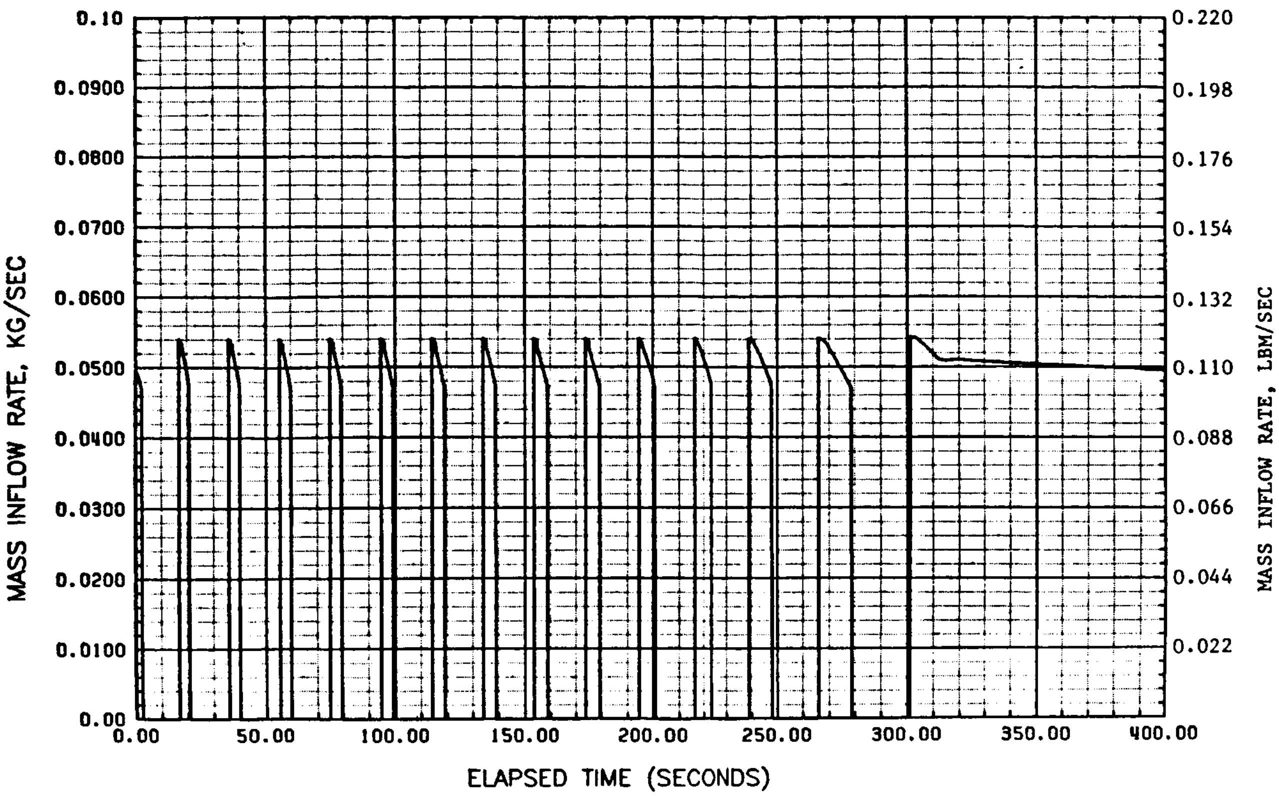


Figure 2-15. MASS INFLOWRATE VS. TIME - LTCFSE RECEIVER TANK CHILLDOWN.

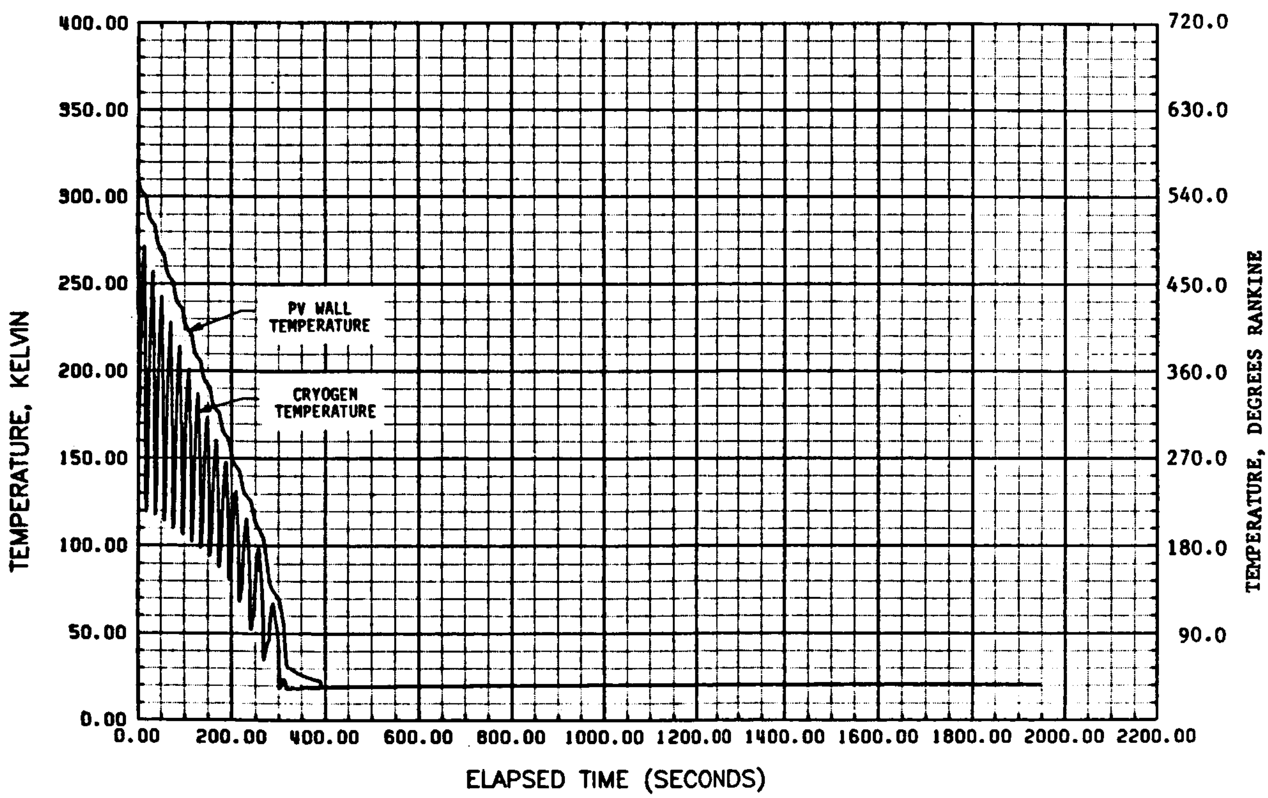


Figure 2-16. TEMPERATURE VS. TIME-LTCFSE RECEIVER TANK CHILLDOWN & FILL.

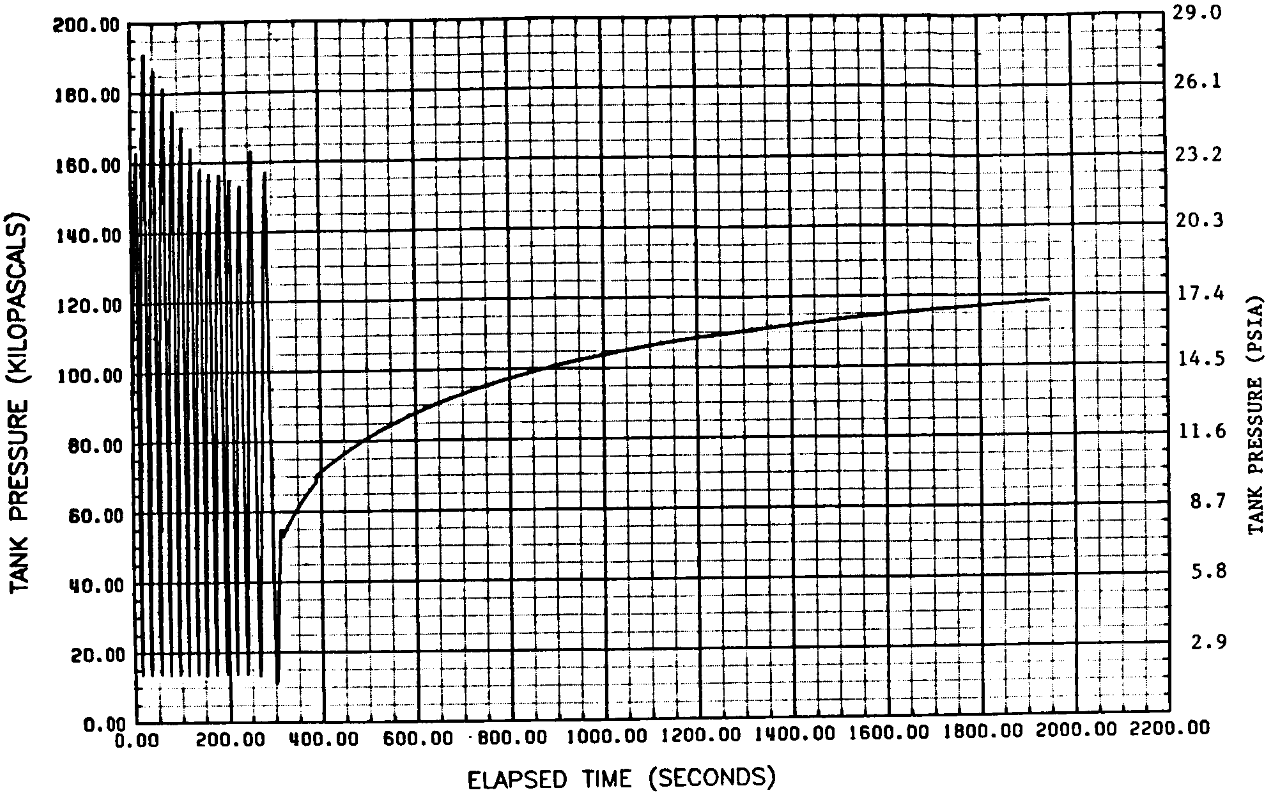


Figure 2-17. PRESSURE VS. TIME - LTCFSE RECEIVER TANK CHILLDOWN & FILL.

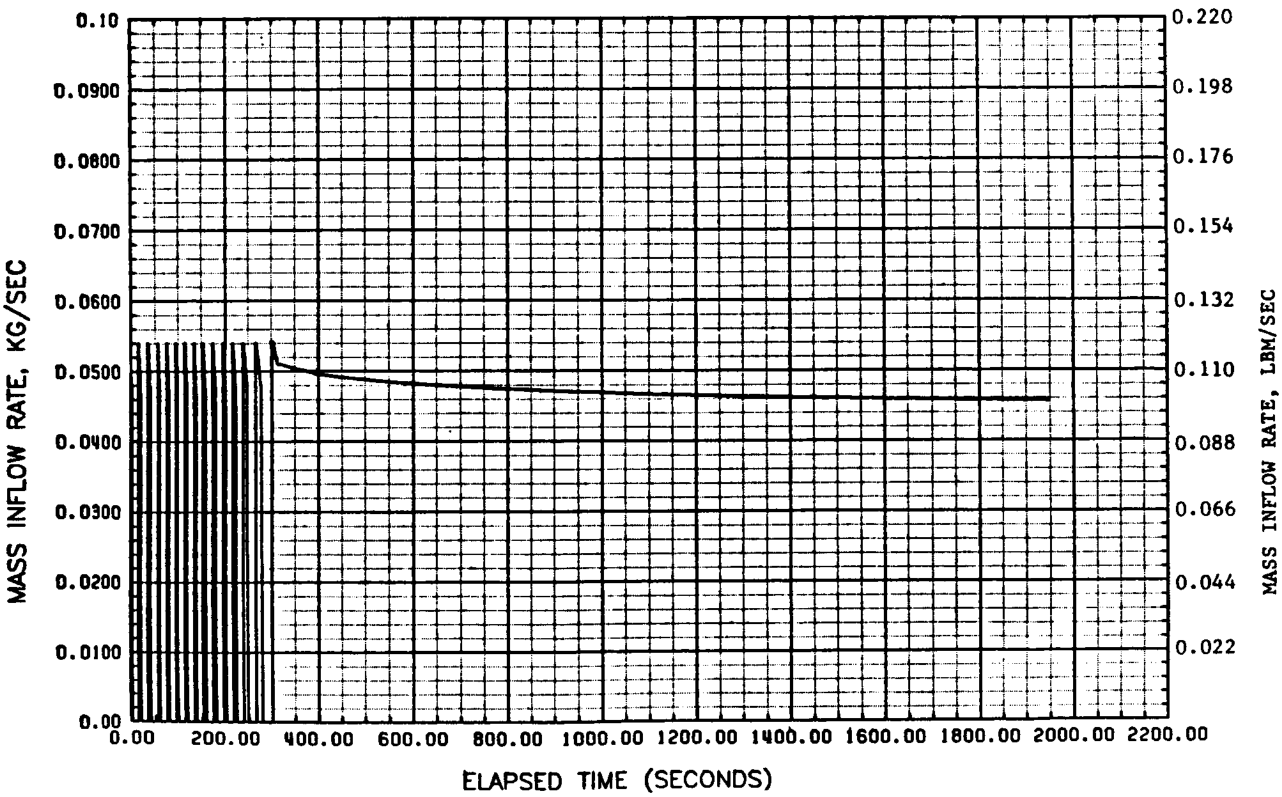


Figure 2-18. MASS INFLOWRATE VS. TIME - LTCFSE RECEIVER TANK CHILLDOWN & FILL.

Figure 2-15 shows the mass inflow rate of cryogen into the receiver tank during the pre-chill cycle. The area under each spike is equal to the total mass injected during each cycle. At 300 seconds, the tank chill/fill begins at a flowrate of .052 kg/sec (.115 lbm/sec) and gradually decreases as tank pressure rises.

Figures 2-16 through 2-18 show the same plots of temperature, pressure and mass inflowrate variation with time, but include both the pre-chill and fill processes. The tank fill lasts from 340 seconds to 2150 seconds. At the end of fill, the cryogen and tank walls are at thermal equilibrium. At termination of fill, the cryogen is saturated at 110 kPa (16 psia).

The TNKCAP program integrates the tank inflow rate to obtain the total mass required for tank cooldown. The total mass of LH_2 required for chilldown was 4.9 kg (10.7 lbm). It is interesting to note the efficiency of allowing the vapor to superheat as opposed to only utilizing the heat of vaporization for cooling as is typically done in one-g cooldown fills. The amount of energy removed from the tank wall during cooldown is 9.03×10^6 J (8560 Btu). The amount of cooling available from 4.9 kg of LH_2 utilizing only heat of vaporization is 2.19×10^6J (2070 Btu). Thus, the cooling capability of the hydrogen is more than four times greater by utilizing the sensible heat of the hydrogen vapor.

2.4.1.3 Fluid Transfer Losses. In addition to tank cooldown losses, losses due to cooldown of the transfer line must also be calculated. The transfer lines must first be sized to calculate these losses. Figure 2-19 presents a plot of transfer line pressure drop vs. flowrate for transfer line diameters ranging from 1.27 to 2.54 cm (0.5 to 1.0 inches) in diameter. The pressure drops shown are for 6.1 m (20 ft) of line, which is the estimated length of Phase II transfer line. A 1.27 cm (0.5 inch) diameter line proves adequate for this purpose. At the nominal tank fill flowrate of 0.045 kg/sec (0.1 lbm/sec), this produces a pressure drop of 6.6 kPa (0.96 psia). Keeping the pressure drop low is desirable to minimize pressurant requirements and the potential for liquid flashing.

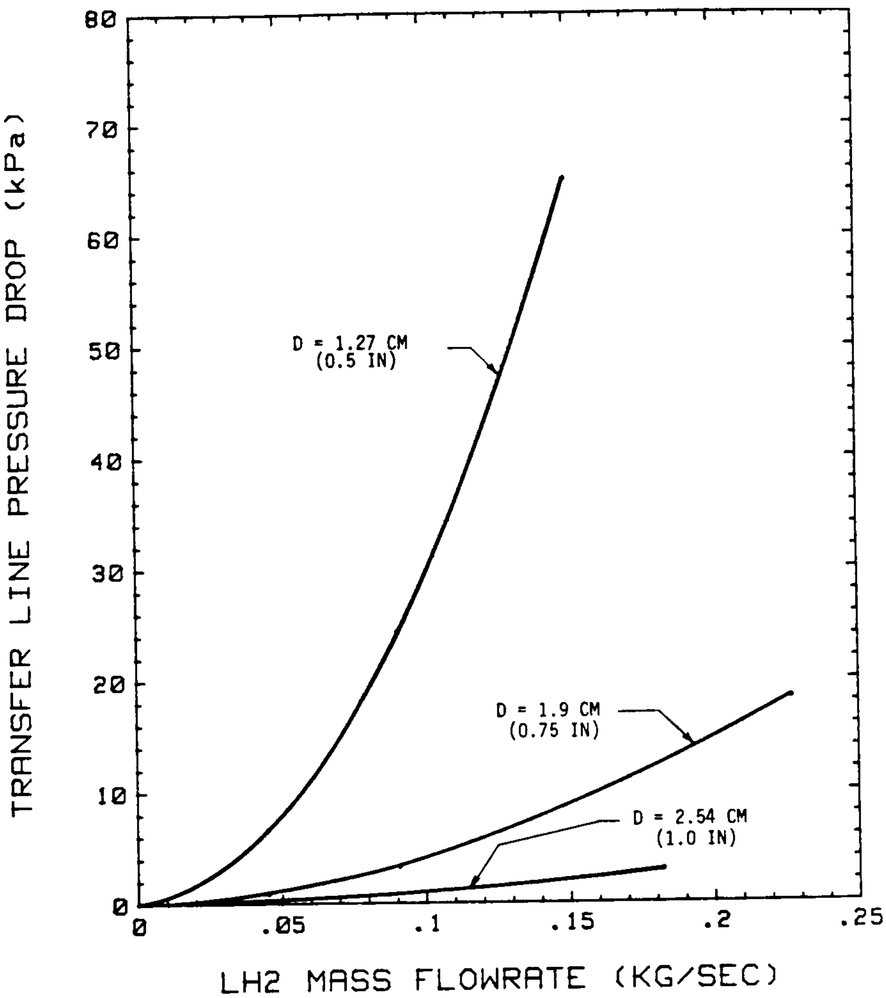


Figure 2-19. TRANSFER LINE PRESSURE DROP VARIATION WITH FLOWRATE.

After the transfer line diameter was sized, the amount of cryogen required for line cooldown was then calculated. To simplify calculation of line cooldown requirements, only the heat of vaporization was assumed to be available for cooling. The transfer line was assumed to be 1.27 cm (0.5 in.) diameter by .89 mm (.035 in) wall stainless steel. The mass of 6.1 m (20 ft) of this line is 1.72 kg (3.8 lbm). A mass of 13.6 kg (30 lbm) was assumed for valves, flow meters and disconnects. These items must also be cooled down prior to fluid transfer. The amount of H_2 required to cool this mass from 300K (540°R) to 22K (40°R) was then calculated. Table 2-XI summarizes the line cooldown fluid requirements for each fluid transfer operation.

Table 2-XI. FLUID TRANSFER COOLDOWN REQUIREMENTS.

ITEM	H_2 MASS REQUIRED	
	kg	(lbm)
Receiver Tank Cooldown	4.9	(10.7)
Transfer Line Cooldown	0.4	(0.8)
Valves, Meter, Disconnects	2.7	(6.1)
TOTAL	8.0	(17.6)

2.4.1.4 Receiver Tank Thermal Performance. Receiver tank thermal performance will be evaluated at the beginning and end of the Phase II test. Each thermal performance test will last 90 days and will evaluate any change in the thermal performance of the soft outer shell receiver tank. Thermal performance of the receiver tank was predicted using the Beech Aircraft Liquid Cryogen Tank Program (Reference 7). The thermal parameters utilized in the analysis and the resulting performance predictions are summarized in Table 2-XII.

Table 2-XII. RECEIVER TANK THERMAL PERFORMANCE.

MLI - 60 layers double aluminized mylar/silk net

MLI emissivity - 0.035

MLI density - 8 layers/cm (20 layers/inch)

Strut suspension system sized for empty PV flight loads

Six struts total A/L - 0.051 cm (0.00167 ft)

Pressurization and fill lines - 1.27 cm x 0.71 mm wall x
127 cm long (0.5" x 0.028" wall x 50" long) 304 Cres

TVS line .48 cm x .71 mm wall (0.1875" x 0.028" wall) 304 Cres

Total line A/L - 4.48×10^{-3} cm (1.47×10^{-4} ft)

Tank heat leak - 1.32 W (4.49 Btu/hr)

Tank boiloff rate - 0.010 kg/hr (0.023 lbm/hr)

Total boiloff during thermal performance testing - 45.1 kg (99.4 lbm)

2.4.1.5 Supply Tank Sizing. With the total Phase II fluid losses determined, an approximate supply tank volume was calculated. A parametric analysis of tank lifetime as a function of volume was performed for a "generic" design LTCFSE supply tank. This design utilized 90 layers of double aluminized mylar with two vapor cooled shields. Dual stage supports were assumed and were sized based on tank suspended weight. The results of this analysis are presented in Figure 2-20 as two curves of tank lifetime versus volume. Tank lifetime is defined as the amount of time required to boiloff all cryogen in the tank. The upper curve in Figure 2-20 is total tank lifetime with no Phase II losses. The lower curve is tank lifetime including Phase II losses. By referring to the Experiment Test Plan, Figure 2-6, a lifetime of approximately four years is required for the LTCFSE supply tank. This corresponds to a tank volume of approximately 5.6 m^3 (198 ft^3), as shown in Figure 2-20.

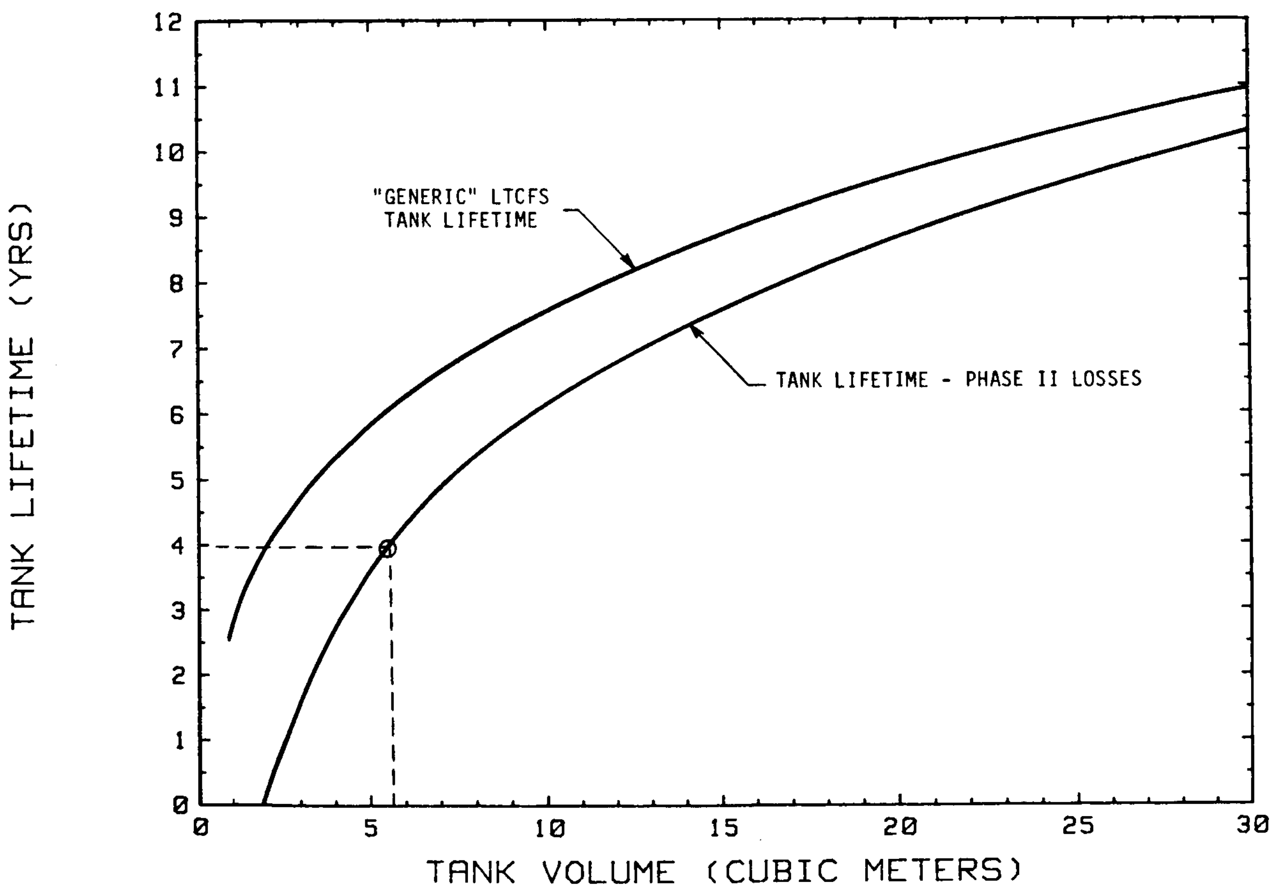


Figure 2-20. LTCFSE SUPPLY TANK LIFETIME VS. VOLUME.

As previously stated, utilization of existing hardware was a primary consideration in the LTCFSE design. Once the approximate supply tank volume was determined, the available hardware was reviewed. This review is summarized in Table 2-XIII. As noted in the table, the Oxygen Thermal Test Article (OTTA) design closely satisfies the LTCFSE supply tank requirements. Details of the OTTA design are included in Appendix C.

Table 2-XIII. HARDWARE SELECTION SUMMARY - TANKS.

ITEM	AVAILABILITY	APPLICABILITY
OTTA	Hardware & design available for supply tank.	Modified OTTA satisfies criteria.
HTTA	Hardware & design available for supply and receiver tank.	Tank volume larger than required.
CFMFE Receiver Tank	Hardware availability questionable - design available.	Too small for supply tank, use as receiver would duplicate CFMFE thermal performance results.
CFMFE Supply Tank	Hardware availability questionable -design available.	Too small for supply tank, m/V too large for receiver tank.
PRSA H_2 Tank	Hardware not available, design available.	Too small for supply tank, m/V too large for receiver.
ELMS Tank	Hardware & design available	Modified ELMS satisfies volume and mass criteria for receiver tank.

2.4.1.6 <u>Pressurization System Selection.</u> Once the supply and receiver tanks had been sized, the final step required to develop the experiment configuration was to select the tank pressurization approach, as shown in Figure 2-10. The baseline system chosen in the Task II preliminary design was an external pressurization loop utilizing an H_2/O_2 gas generator. The gas generator was coupled to a heat exchanger that conditioned hydrogen from the supply tank to a slightly superheated state, then reinjected it into the supply tank for pressurization. A major problem with this system is that the H_2/O_2 gas generator exhausts H_2O vapor. H_2O contamination is undesirable in the vicinity of Space Station as it absorbs several important frequencies of electromagnetic radiation, particularly in the infrared region. Six alternate pressurization systems were compared to the pressurization system chosen during Task II. Weight, volume, resource requirements, and contamination and safety issues were reviewed and compared to the Task II baselined system. These systems are summarized in the following paragraphs.

System No. 1 - Hydride Boiloff Collection - No Accumulator. This system, shown schematically in Figure 2-21 utilizes dual 0.03 m^3 (1 ft^3) $LaNi_5$ metal hydride beds to collect and store supply tank boiloff. Hydrogen is then expelled from the beds to provide the pressurization required during fluid transfer operations.

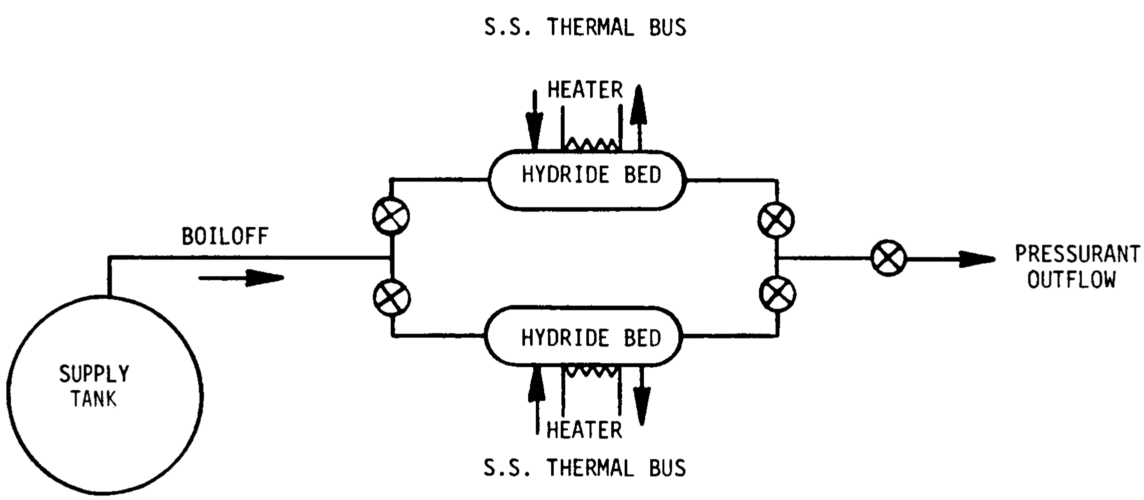


Figure 2-21. PRESSURIZATION SYSTEM NO. 1 SCHEMATIC.

Metal hydrides are materials that absorb hydrogen in an exothermic reaction, storing the hydrogen at densities approximating that of the liquid storage. The hydrogen may be stored indefinitely and later expelled by applying heat to the hydride bed. References 8 and 9 contain further information on hydrogen storage utilizing metal hydrides.

Referring to Figure 2-21, the system functions as follows: boiloff gases are collected in one hydride bed, which is cooled by the Space Station thermal bus. The cooling is necessary to remove the heat of reaction produced during the exothermic absorption process. The Space Station thermal bus will supply 14.6 W (50 Btu/hr) of cooling during absorption of boiloff gases. Two hydride beds are utilized so boiloff may still be collected during tank pressurization. Each hydride bed will store enough hydrogen for one complete transfer operation. When the first bed is full, the second bed will begin absorption. At this point, the first bed may be used for pressurization. Two kilowatts of power is required to provide the needed pressurant flowrate GH_2. The advantages and disadvantages of this system are summarized in Table 2-XIV.

Table 2-XIV. PRESSURIZATION SYSTEM NO. 1 - ADVANTAGES AND DISADVANTAGES.

ADVANTAGES	DISADVANTAGES
o Lowest volume hydride system o Low cooling requirements o Low maximum pressure 345 kPa (50 psia) o Conserves cryogen relative to Systems No. 4 and 7	o High electrical power requirements (2 kW) for expulsion o Cannot be used to collect tank cooldown gases o High mass

System No. 2 - Hydride Boiloff Collection With Accumulator. This system, shown in Figure 2-22 is operationally very similar to System No. 1. The major difference is that when the first bed is full, it expells hydrogen at a low flowrate to a 0.6 m^3 (21 ft^3), 3.45 MPa (500 psia) accumulator. This system uses the hydride beds as a GH_2 compressor. While one bed is being cooled and is absorbing GH_2 boiloff gases, the other bed is expelling GH_2 to the accumulator. When the accumulator is full, the hydrogen is either used for tank pressurization, or it is transferred to the Space Station for use. Use of the accumulator allows the hydride bed to be smaller (.003 m^3, 0.1 ft^3) than in System No. 1 and reduces system mass and expulsion power requirements. The advantages and disadvantages of this system are summarized in Table 2-XV.

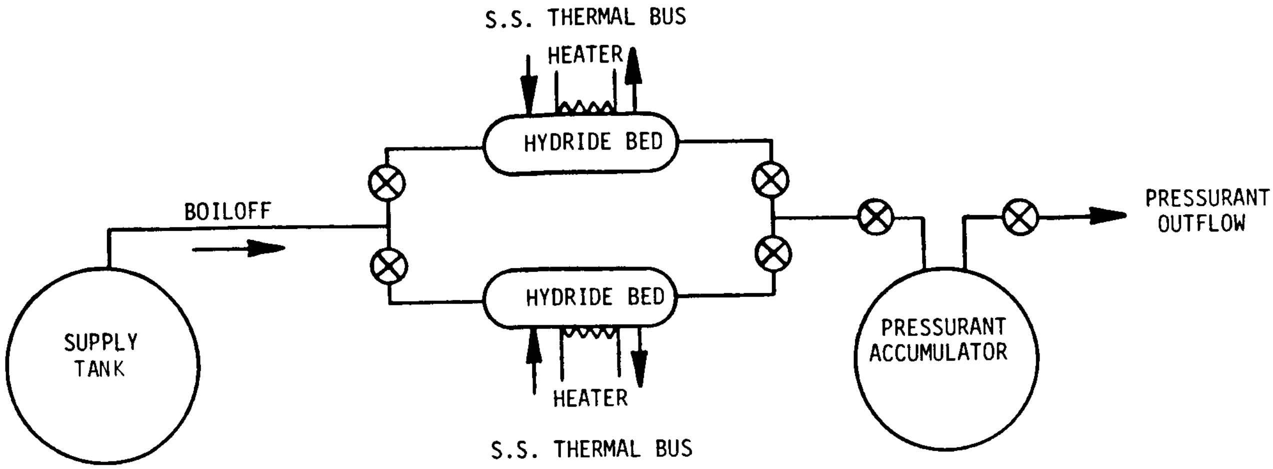


Figure 2-22. PRESSURIZATION SYSTEM NO. 2 SCHEMATIC.

Table 2-XV. PRESSURIZATION SYSTEM NO. 2 - ADVANTAGES AND DISADVANTAGES.

ADVANTAGES	DISADVANTAGES
o Lowest mass system o Low electrical and cooling requirements o Conserves cryogen relative to Systems No. 4 and 7	o High volume o Cannot be used to collect cooldown gases

System No. 3 - Hydride Boiloff and Cooldown Gas Collection. System No. 3, depicted in Figure 2-23, is operationally identical to System No. 1. The hydride beds in this system are 0.085 m^3 (3 ft^3) each and have been sized to allow collection of the receiver tank cooldown gases in addition to tank boiloff. This completely eliminates experiment venting. However, 8 kW of cooling is required during collection of cooldown gases due to the high GH_2 vent flowrates. The advantages and disadvantages of this system are summarized in Table 2-XVI.

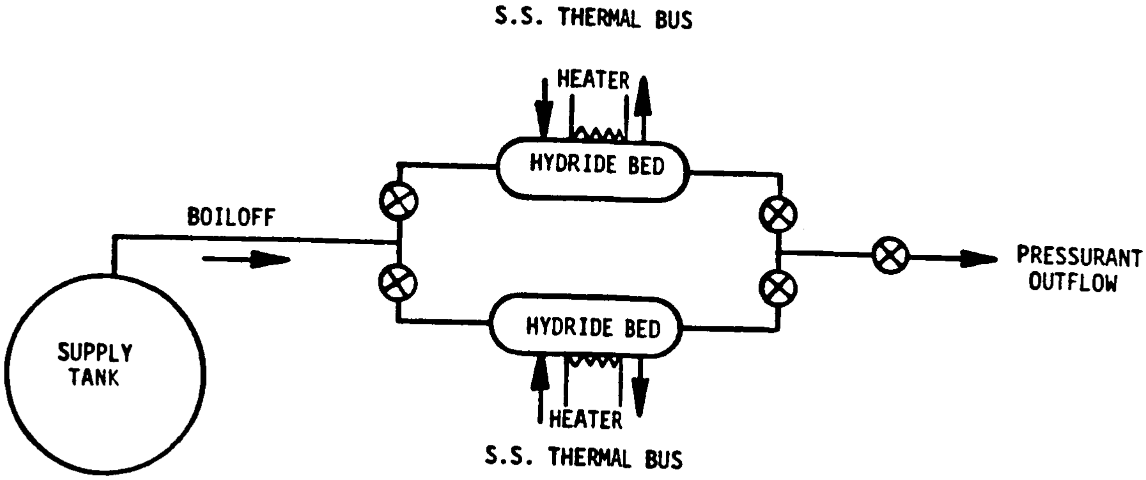


Figure 2-23. PRESSURIZATION SYSTEM NO. 3 SCHEMATIC.

Table 2-XVI. PRESSURIZATION SYSTEM NO. 3 - ADVANTAGES AND DISADVANTAGES.

ADVANTAGES	DISADVANTAGES
o No experiment venting o Conserves cryogen relative to Systems No. 4 and 7	o Highest mass system o High cooling and power requirements

<u>System No. 4 - High Pressure Gas.</u> This system, depicted in Figure 2-24, is a simple high pressure gas bottle system. A 1.53 m^3 (54 ft^3), 20.7 MPa (3000 psia) Kevlar wrapped aluminum gas bottle is utilized to store enough GH_2 to perform the ten transfer operations required in Phase II. Although this system is operationally simple, it has a large mass and volume since pressurant for all transfer operations must be stored. The advantages and disadvantages of this system are summarized in Table 2-XVII.

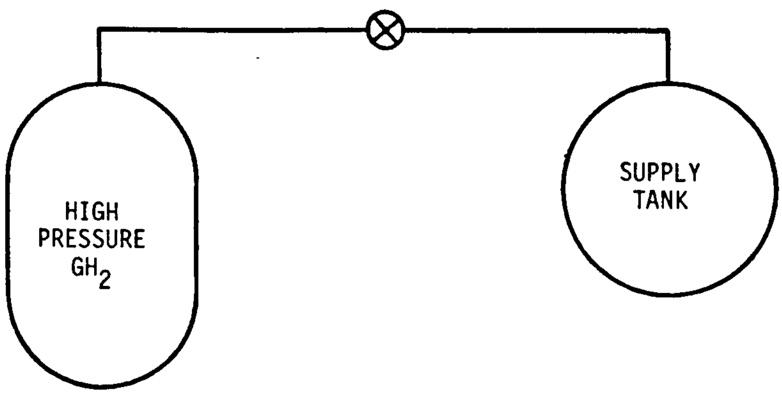


Figure 2-24. PRESSURIZATION SYSTEM NO. 4 SCHEMATIC.

Table 2-XVII. PRESSURIZATION SYSTEM NO. 4 - ADVANTAGES AND DISADVANTAGES.

ADVANTAGES	DISADVANTAGES
o Utilizes simple, well developed technology o Operationally simple o No Space Station resources required o No time constraint between pressurizations	o Highest volume system o No new technology demonstration gained from use o Safety hazard due to high pressure o Experiment continually vents GH_2

System No. 5 - Boiloff Collection with Compressor and Accumulator. System No. 5, depicted in Figure 2-25, utilizes a mechanical compressor to collect boiloff gases and stores the pressurized boiloff in a 0.16 m^3 (5.5 ft^3), 20.7 MPa (3000 psia) Kevlar wrapped composite bottle. When the accumulator is completely charged, enough pressurant is available for one fluid transfer operation. The advantages and disadvantages of this system are summarized in Table 2-XVIII.

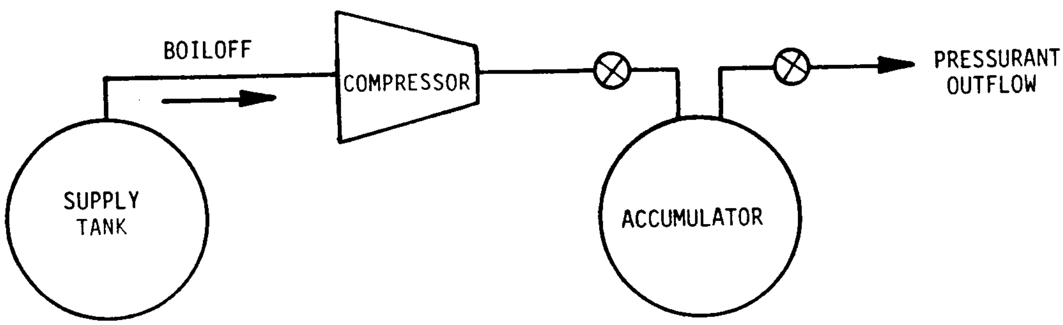


Figure 2-25. PRESSURIZATION SYSTEM NO. 5 - SCHEMATIC.

Table 2-XVIII. PRESSURIZATION SYSTEM NO. 5 - ADVANTAGES AND DISADVANTAGES.

ADVANTAGES	DISADVANTAGES
o Simple operation o Fewer Space Station interfaces than hydride system o Simple, well-developed technology	o Does not collect tank cooldown gases o Requires reliable, long lifetime compressor with backup

System No. 6 - Boiloff and Cooldown Gas Collection with Compressor and Accumulator. This system, shown in Figure 2-26, is operationally similar to System No. 5. An additional compressor and accumulator have been added to collect the receiver tank cooldown gases. The second compressor is required due to the much higher mass flowrate of the cooldown gases. The additional 0.47 m^3 (16.5 ft^3) accumulator is used to store the cooldown gases from one tank cooldown. Alternatively, the two accumulators could be combined into one 0.62 m^3 (22 ft^3) accumulator; however, this does not change the overall system mass appreciably. The advantages and disadvantages of this system are summarized in Table 2-XIX.

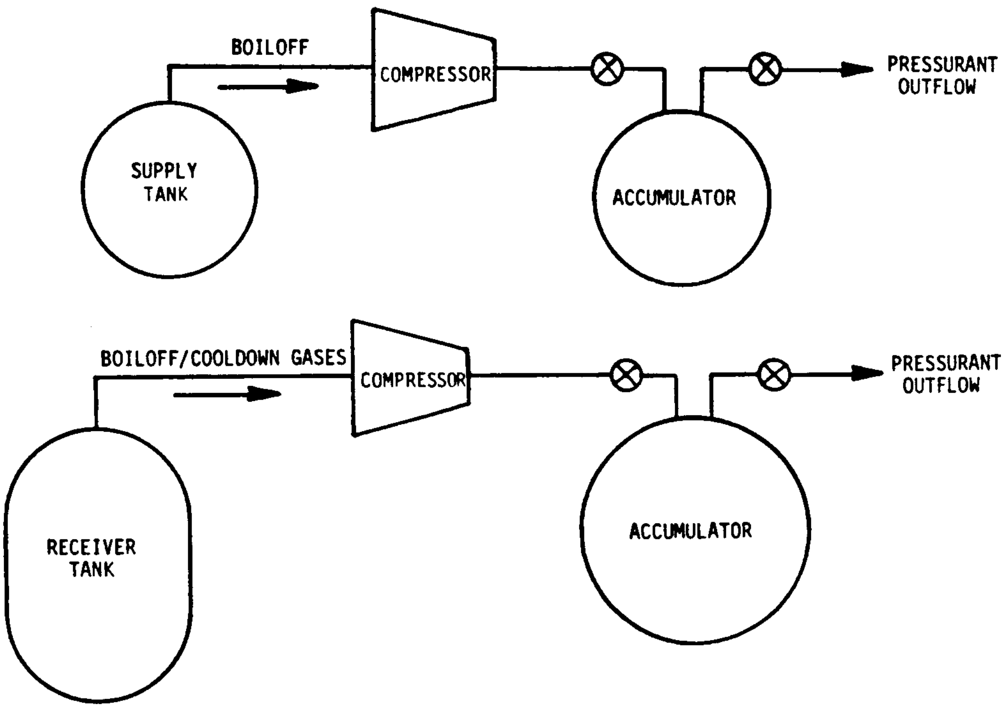


Figure 2-26. PRESSURIZATION SYSTEM NO. 6 - SCHEMATIC.

Table 2-XIX. PRESSURIZATION SYSTEM NO. 6 - ADVANTAGES AND DISADVANTAGES.

ADVANTAGES	DISADVANTAGES
o Experiment never vents GH_2 o Simple operation o Fewer Space Station interfaces than hydride systems o Simple, well-developed technology	o High weight and volume o High power requirements during tank cooldown o Requires long lifetime, reliable compressors with backups

System No. 7 - External Pressurization Loop with Gas Generator and Heat Exchanger. This system, depicted in Figure 2-27 utilizes a gas generator and heat exchanger to condition LH_2 drawn from the supply tank to a slightly superheated state. This superheated vapor is then utilized for pressurant. Liquid hydrogen is pumped from the supply tank and the flow is then split, part of it going to the heat exchanger for conditioning, and the remainder is combined with GO_2 from a high pressure bottle and combusted in the gas generator. The hot combustion products are passed over the heat exchanger to condition the hydrogen and then exhausted. The advantages and disadvantages of this system are summarized in Table 2-XX. This was the pressurization system baselined in the Task II preliminary design.

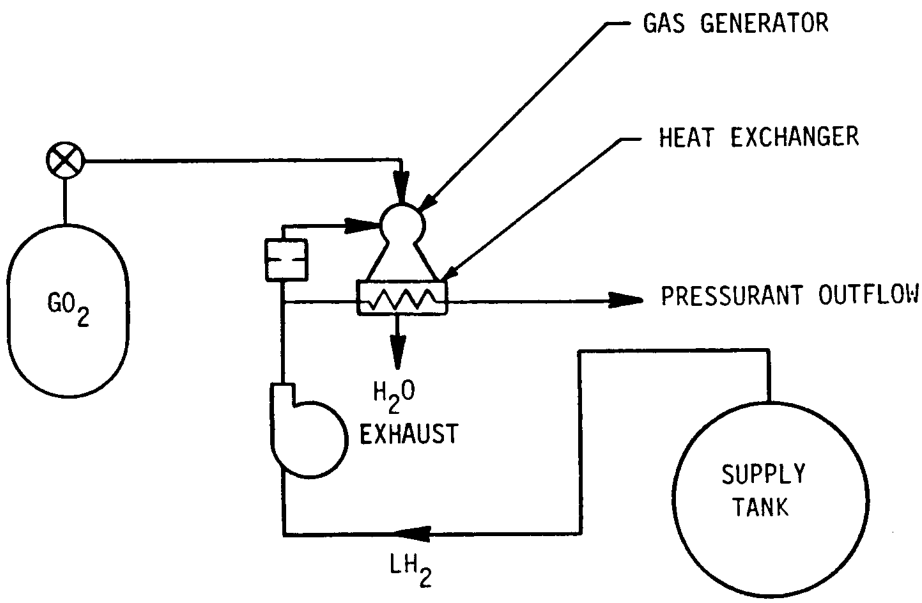


Figure 2-27. PRESSURIZATION SYSTEM NO. 7 - SCHEMATIC.

Table 2-XX. PRESSURIZATION SYSTEM NO. 7 - ADVANTAGES AND DISADVANTAGES.

ADVANTAGES	DISADVANTAGES
o High expulsion rate o Demonstrates new technology o Low mass o Moderate power consumption o Allows for complete supply tank expulsion at any time for contingencies	o Exhausts H_2O vapor o More system safety issues than other systems o Experiment vents GH_2 continuously o Depletes LH_2 supply for pressurization

Pressurization System Trade Study A summary of the pressurization system trade study is presented in Table 2-XXI. This table outlines system weights, volumes, and resource requirements for the seven systems investigated. It should be noted that systems one through six all utilize superheated ortho hydrogen for pressurant. Tank boiloff will increase as the pressurant reaches thermal equilibrium with the saturated tank fluid and converts back to para hydrogen. Preliminary calculations indicate that approximately 10 kg (22 lbm) of additional hydrogen will boiloff due to this effect. The supply tank has an adequate margin of additional H_2 mass to satisfy this requirement. Based on these parameters, and the comparisons of Tables 2-XIV through 2-XX, System No. 2, Hydride Boiloff Collection with Accumulator, was chosen. This system minimizes mass, resource requirements, and safety considerations. It is highly reliable and will also provide a new technology demonstration.

Table 2-XXI. PRESSURIZATION SYSTEM TRADE STUDY RESULTS.

OPTION	WEIGHT kg (lbm)	POWER REQUIRED (WATTS)[1]	COOLING REQUIRED (WATTS)	SYSTEM VOLUME m^3 $(ft^3)^2$
1) Hydride for B/O Collection, no accumulator	454 (1000)	2000 (during expulsion only)	15	0.057 (2.0)
2) Hydride for B/O Collection with accumulator	91 (200)	15	15	0.6 (21.2)
3) Hydride, no accumulator, collects boiloff and cooldown gases	1361 (3000)	2000 (during expulsion)	8 Kw (during receiver tank cooldown only)	0.17 (6.0)
4) High pressure gas	272 (600)	--	--	1.53 (54.0)
5) Compressor with accumulator to collect boiloff only	118 (260)	30	--	0.18 (6.5)
6) Dual compressors and accumulators to collect boiloff and cooldown gases	481 (1060)	2700 (during cooldown)	--	0.79 (28.0)
7) External pressurization loop with gas generator and heat exchanger	181 (400)	100	--	0.14 (5.0)

1 Does not include instrumentation and control power
2 Does not include valve and line volume

Figure 2-28 presents a schematic of this system that illustrates how system control will be achieved. As a hydride bed is heated, H_2 is expelled, increasing the hydride bed pressure. Conversely as it is cooled, H_2 is absorbed, lowering the pressure. This principle allows control of the system to be achieved utilizing check valves rather than solenoid operated valves. As depicted in the figure, a check valve module is located at each end of both hydride beds. Each module consists of four individual check valves in a series-parallel arrangement. This allows for proper valve operation in the event of a check valve failure in either the closed or open mode.

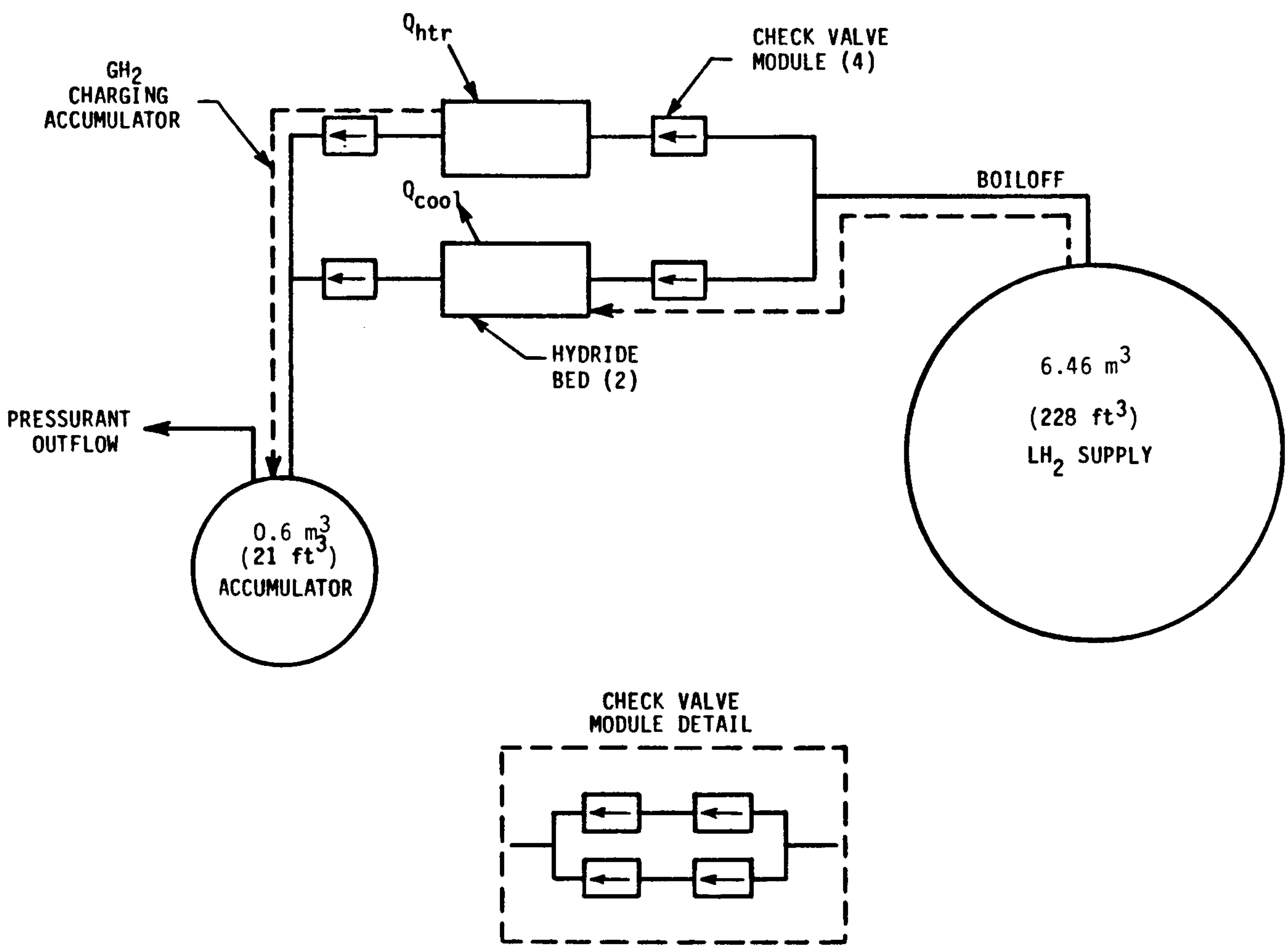


Figure 2-28. PRESSURIZATION SYSTEM OPERATION.

As shown in the figure, the lower hydride bed is being cooled. This will lower the pressure within the bed, causing the check valve between it and the supply dewar to open, thereby allowing boiloff gas to enter the hydride bed and be absorbed. The check valve between the accumulator and lower hydride bed remains closed because the accumulator pressure is higher than the lower hydride bed pressure. The upper hydride bed, which has been previously charged with H_2, is being heated, thus increasing its pressure. This increase in pressure keeps the check valve between the upper bed and the supply dewar closed. As the upper hydride bed increases to a pressure higher than that in the accumulator, the check valve between the upper hydride bed and the accumulator opens, expelling H_2 into the accumulator. This process continues until the upper hydride bed is depleted and the lower hydride bed is completely charged. At this point, the heating and cooling cycles are reversed, and the process continues until the accumulator is charged to 3.45 MPa (500 psia). Thus, the pressurization system can be controlled merely by alternately heating and cooling the hydride beds.

2.4.2 Detailed Conceptual Design. Based on the configuration development studies, a detailed conceptual design was performed for each phase of the experiment. This design is described in detail in the following section, and includes:

1. Configuration drawings and descriptions
2. Equipment list
3. Instrumentation list
4. Space Station interface and resource requirements
5. System schematics
6. System deployment and operations descriptions

2.4.2.1 Phase I Description. Phase I of the experiment is designed to demonstrate basic passive thermal control technologies. In addition, hardware necessary for interfacing with Phases II and III is included. An isometric view of the Phase I hardware is presented in Figure 2-29. Three view drawings of the configuration are presented in Figures 2-30 through 2-32. The Phase I configuration consists of a 6.46 m^3 (228 ft^3) LH_2 supply dewar mounted within an aluminum support structure. Standard trunnion pin mounts are used as framework mounting interfaces for STS launch and Space Station deployment. Both keel and payload bay bridge fitting trunnions are utilized. Fluid and electrical interface panels are provided for the Space Station and Phases II and III interfaces. A high pressure GHe bottle is provided for STS contingency dump.

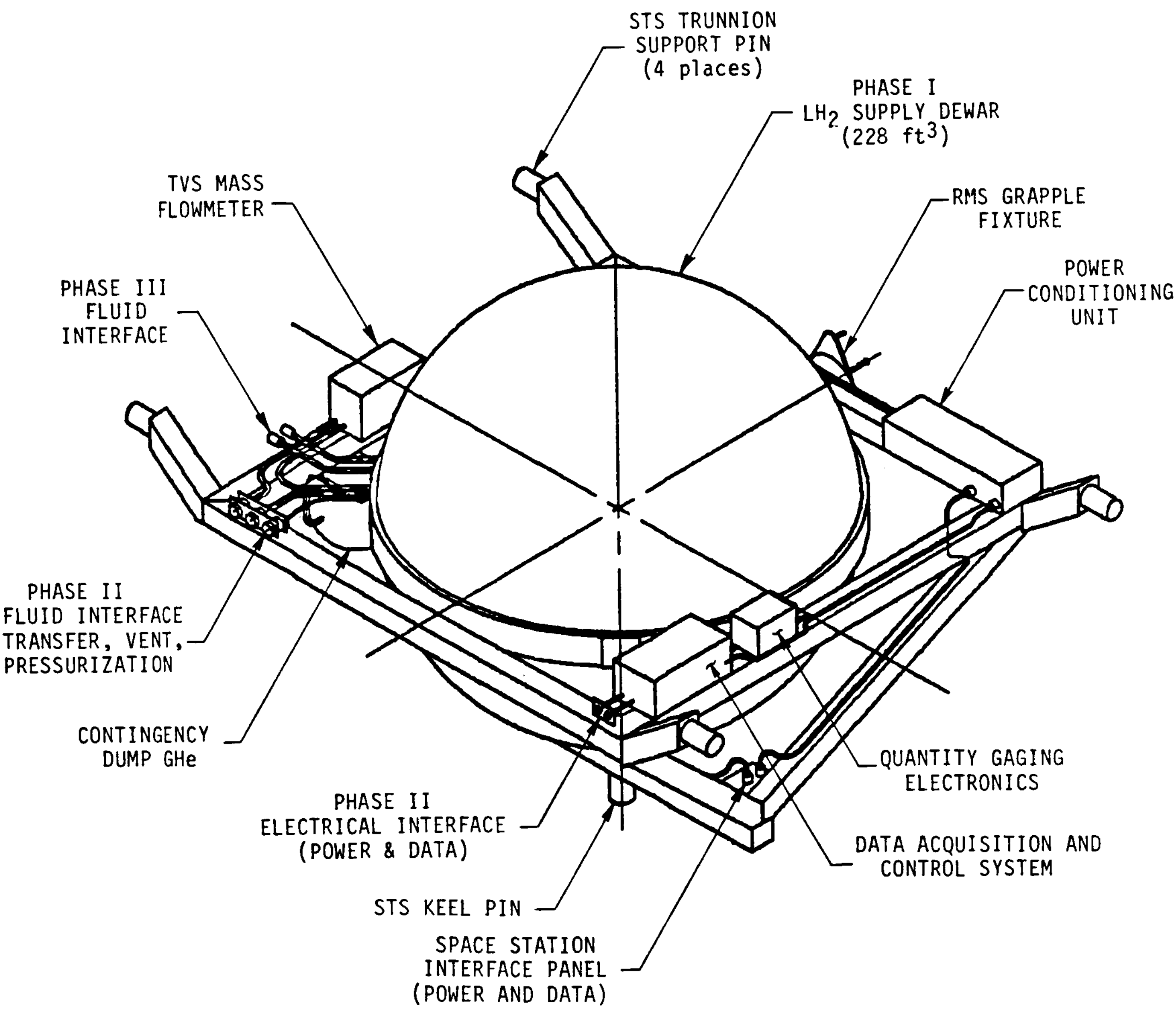


Figure 2-29. PHASE I CONFIGURATION - ISOMETRIC VIEW.

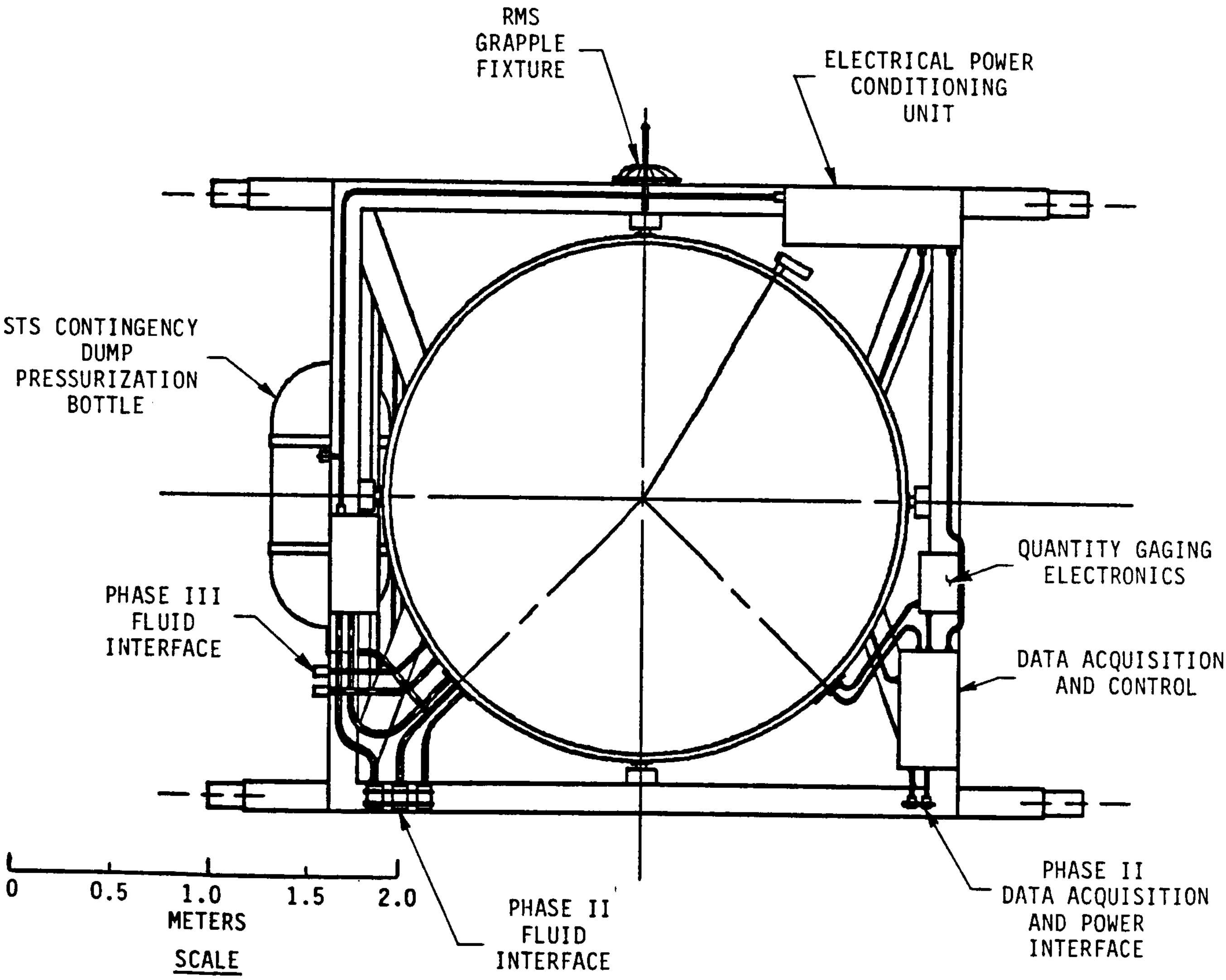


Figure 2-30. PHASE I CONFIGURATION - TOP VIEW.

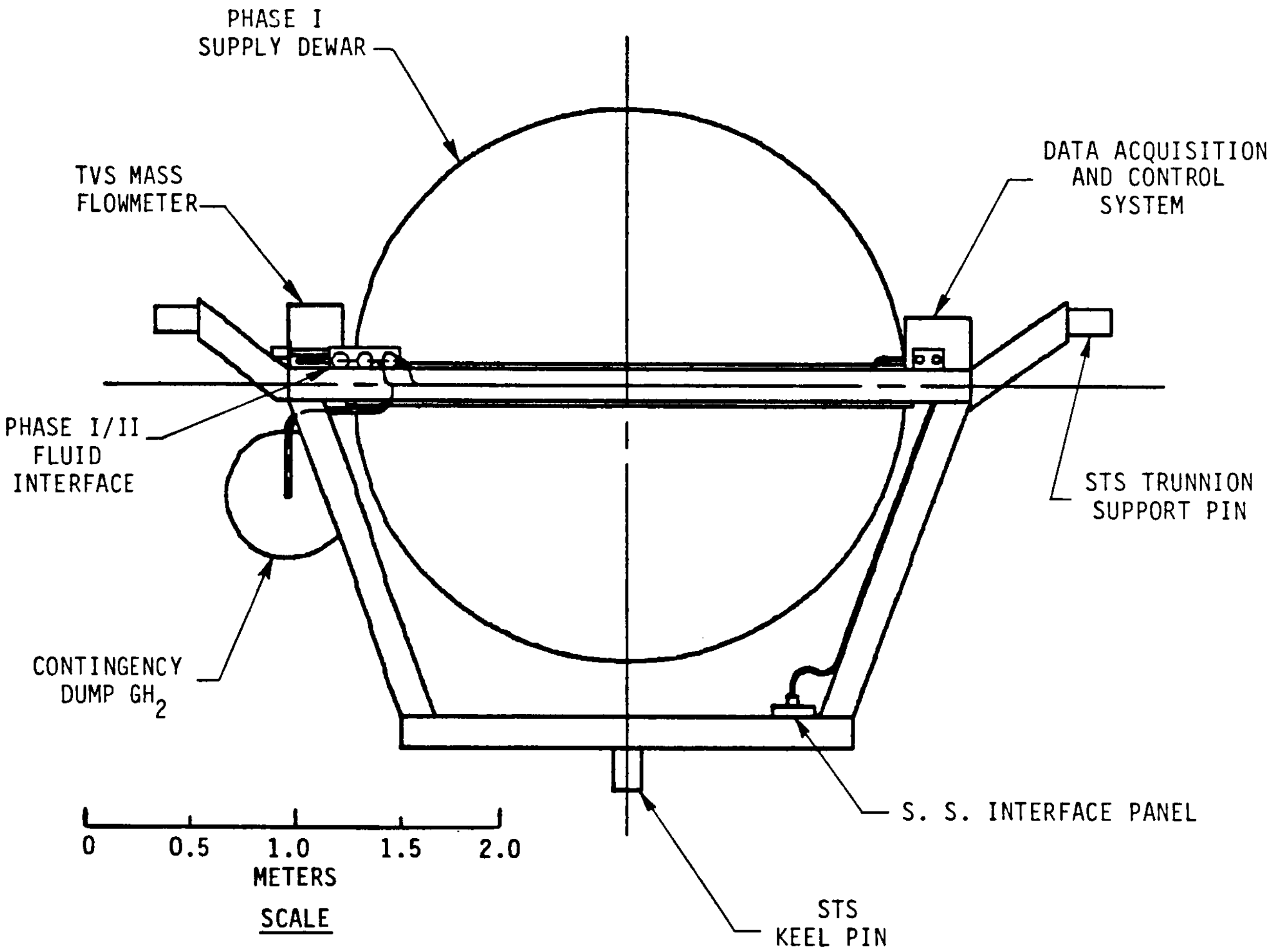


Figure 2-31. PHASE I CONFIGURATION - FRONT VIEW.

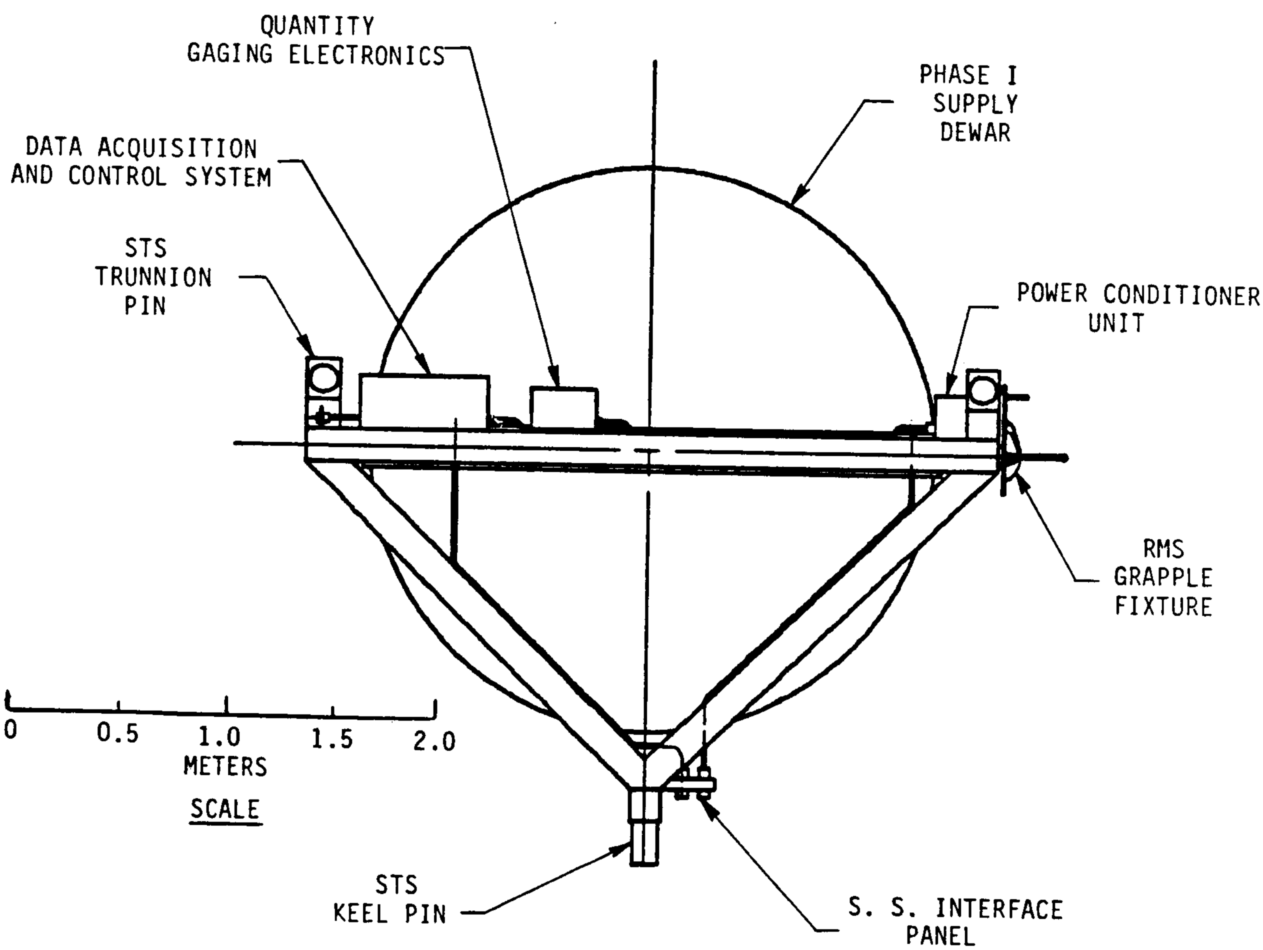


Figure 2-32. PHASE I CONFIGURATION - SIDE VIEW.

Electronic "black boxes" are mounted on the support structure for data acquisition and control purposes. The following systems will be required and are described in detail below:

1. Data acquisition and control
2. Power conditioning unit
3. Mass flow meter
4. Low-g quantity gaging electronics

The Space Station configuration is currently baselined as having an on-board data acquisition system available for use by attached payloads. It is recommended this system be utilized by the experiment, as this removes the cost of developing an independent data acquisition system. When the Space Station data acquisition system becomes defined in sufficient detail, a list of sensors compatible with the system will be issued to potential users. Utilizing these sensors will allow a direct interface with the data acquisition system, minimizing cost and complexity of the experiment data acquisition hardware.

An additional service, named Telescience, will also be available for Space Station users. This service will allow a real time link between the user on the ground and experiments aboard the Space Station. This will be accomplished via dial-up computer lines and an RF link to Space Station via the Tracking and Data Relay Satellite (TDRS) System. This link will allow users to access data on a real-time basis, and to change parameters such as data sampling rates. Use of this data acquisition system minimizes data acquisition costs and provides the experiment with a versatile and powerful data acquisition capability.

Use of the Space Station data acquisition system simplifies the requirements of the LTCFSE experiment Data Acquisition and Control System (DACS) by minimizing the need for on-board signal conditioning and data storage. A block diagram of the LTCFSE experiment DACS is presented in Figure 2-33. Instrumentation signals pass through a signal conditioner, analog multiplexer and analog/digital converter, if required. However, in most cases, utilization of sensors compatible with the Space Station data acquisition system will allow direct connection of sensors to the Space Station data interface. The DACS Central Processing Unit (CPU) will utilize control algorithms stored in the Read-Only-Memory (ROM), or real-time commands from the Space Station (via the telemetry interface), to control the experiment via the Digital/Analog Converter. Temporary data storage can be accommodated using the Random Access Memory (RAM). The DACS also contains redundant processors and memory fault detection capabilities.

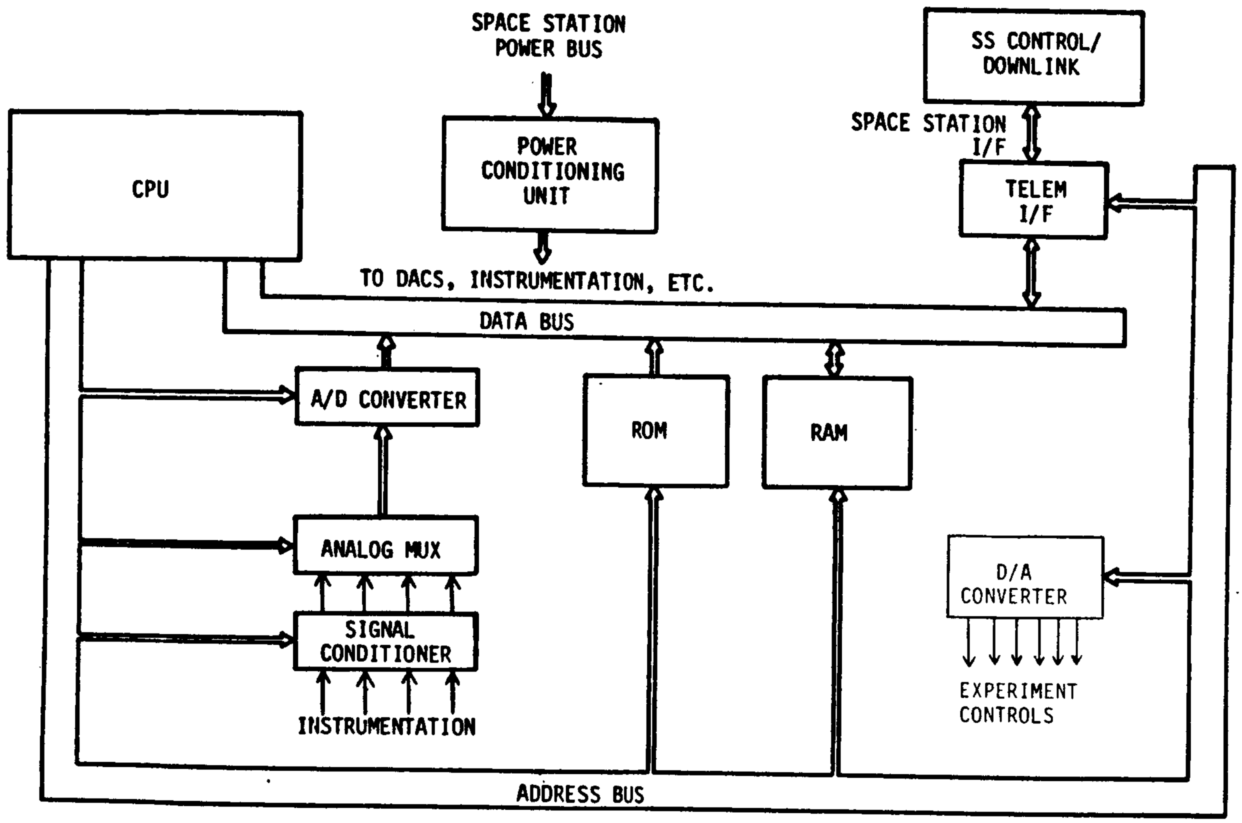


Figure 2-33. BLOCK DIAGRAM - DATA ACQUISITION AND CONTROL SYSTEM.

The Power Conditioning Unit (PCU) is a separate module that provides power to all electronic modules, using the Space Station electrical power bus as a power source. Power interfaces are provided for the Phase II module as well. Due to the much higher Phase III power requirement, a separate PCU will be provided on the Phase III module. Should the DACS fail, mechanical backup devices, such as pressure relief valves and burst discs, are provided in all systems to ensure catastrophic failure will not occur. Each solenoid valve in the system will actually be a valve module of four valves, in a series parallel arrangement. This allows proper system operation if a valve failure occurs in either closed or open mode.

The mass flowmeter will measure TVS boiloff to determine Phase I supply tank thermal performance. A specific type of mass flowmeter will be selected when details of the Space Station data acquisition system are defined.

The low-g quantity gaging system baselined for the LTCFSE experiment is a radio frequency modal analysis quantity gage. This type of system is currently in development by Johnson Space Center. It utilizes standing wave electromagnetic field patterns generated by an antenna inside the tank to determine cryogen quantity. These electromagnetic wave patterns occur at resonant mode frequencies which are dependent on the mass of cryogen present in the tank. By determining the ratio of resonant frequencies for a given mode between a tank empty state (determined during calibration) and the state being measured, cryogen quantity may be determined. Figure 2-34 depicts a block diagram of an RF quantity gaging system. The antenna utilized for this sytem is mounted inside the tank and is shown in Figure 2-35.

A cut-away view of the Phase I supply dewar showing details of the TVS is presented in Figure 2-36. Liquid vented via the LAD is throttled through the Joule-Thomson valve, partially vaporizing the liquid and lowering its temperature. This liquid vapor mixture is then passed through the pressure vessel wall heat exchanger to reduce heat leak into the pressure vessel. The PV wall heat exchange tubing will be routed near areas where localized heat leaks occur, such as at strut and vent line interfaces, in order to intercept as much of these heat leaks as possible. Figure 2-37 illustrates the TVS line routed near a strut interface, as well as the MLI wrapping technique that will be utilized at this interface to minimize radiation heat leak. After exiting the PV wall heat exchanger, the fluid flows through an inner VCS.

At the exit of the inner VCS, the fluid flows through a para-to-ortho converter, lowering the fluid temperature to provide further cooling. The fluid then passes through heat stationing points to reduce fluid line and strut conduction heat leak and then through the outer VCS prior to exiting the tank.

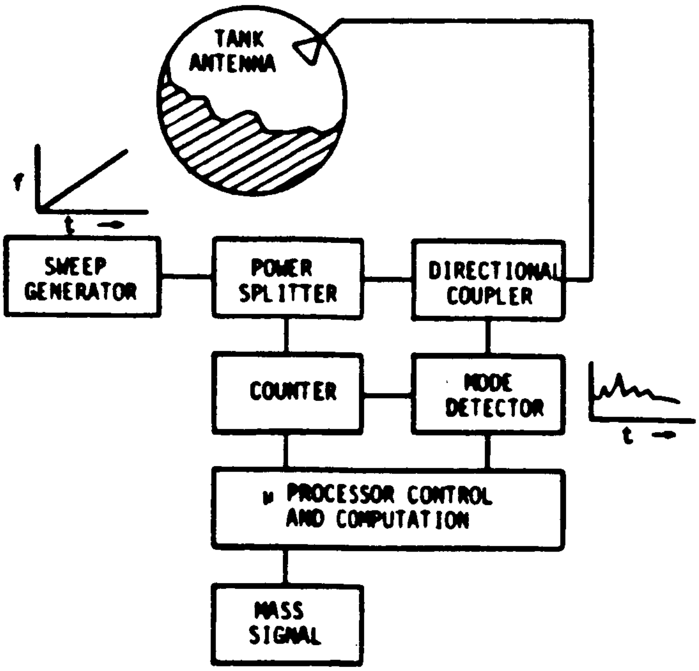
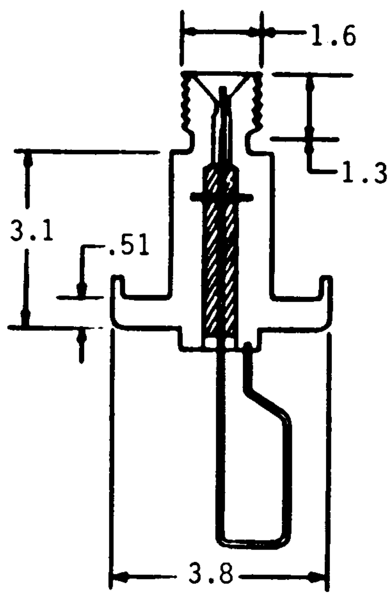


Figure 2-34. RF QUANTITY GAGING SYSTEM BLOCK DIAGRAM.



NOTE: ALL DIMENSIONS IN cm

Figure 2-35. RF QUANTITY GAGING SYSTEM ANTENNA.

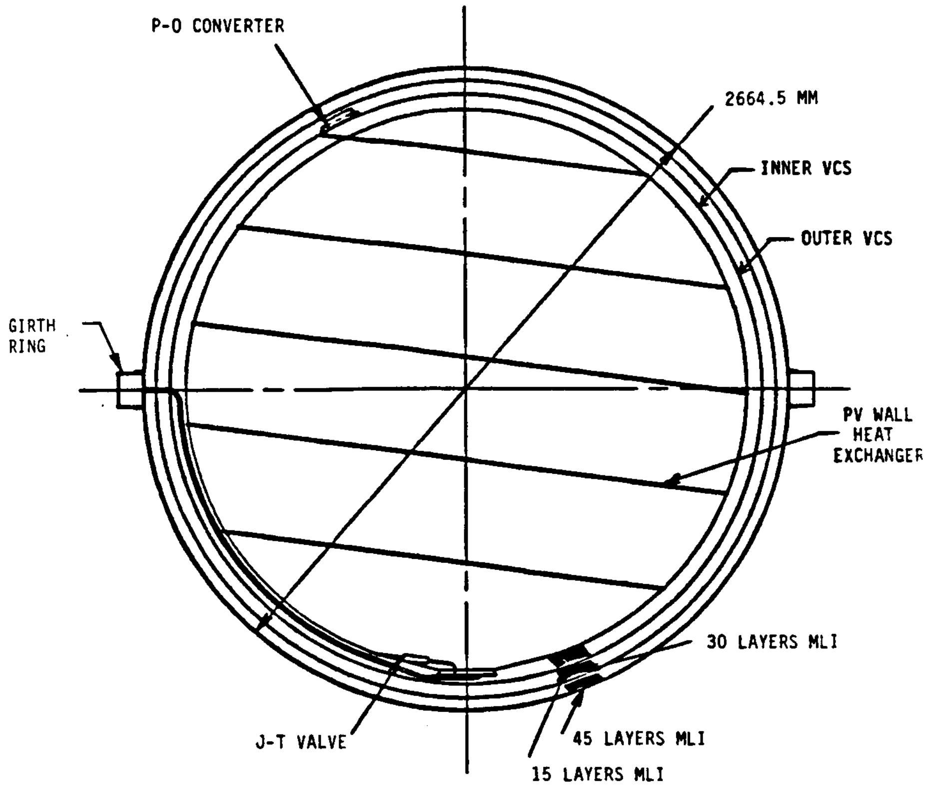


Figure 2-36. SUPPLY DEWAR TVS DETAIL.

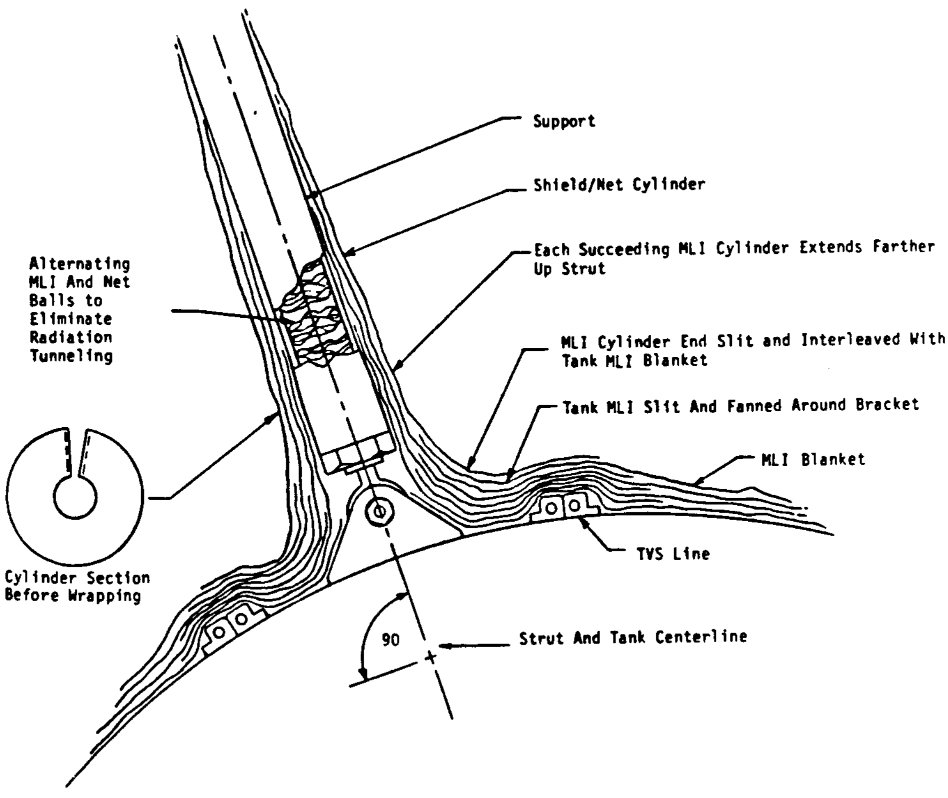


Figure 2-37. STRUT/PV INTERFACE DETAIL.

The tank insulation system consists of 90 layers of Double Aluminized Mylar (DAM) with an emissivity of 0.035. The insulation system utilizes silk netting between layers to reduce conduction heat leak and is installed at a density of 8 layers/cm (20 layers/inch). The distribution of MLI layers, as shown in Figure 2-36 is designed to provide minimal heat leak for the 90 layer two VCS configuration.

The para-to-ortho H_2 converter is shown in detail in Figure 2-38. Parahydrogen vapor from the inner VCS enters the converter and flows radially outward through the catalyst bed. The parahydrogen is cooled as conversion to the equilibrium mixture occurs. The equilibrium hydrogen exits the converter and is routed to heat stationing points and the outer VCS.

A cut-away view of the pressure vessel showing the LAD is presented in Figure 2-39. The LAD consists of four channels at 90° intervals. The inner surface of each channel contains a stainless steel fine mesh screen to acquire and contain cryogen. The maximum flowrate through the LAD that occurs during an on-orbit abort will drive the LAD size. A ground vent line is routed from the top of the tank to the LAD exit to provide vapor venting during ground servicing and fill operations. In order to minimize the possibility of boiling occurring within the LAD, it will be thermally coupled to the PV wall heat exchanger.

A description of the Phase I supply tank, along with thermal performance characteristics, is presented in Table 2-XXII. The supply tank thermal performance is compared to the unmodified OTTA thermal performance in Figure 2-40. The supply tank performance, labeled "Modified LTCFSE OTTA" is shown on a parametric performance versus volume line. The OTTA data point is based on ground test data from testing performed at the Beech Aircraft Boulder Division. The 20% decrease in heat leak predicted is primarily due to dual stage supports, thick MLI and outer shell thermal coatings. This comparison to ground test data indicates that the projected LTCFSE supply dewar thermal performance is quite achievable.

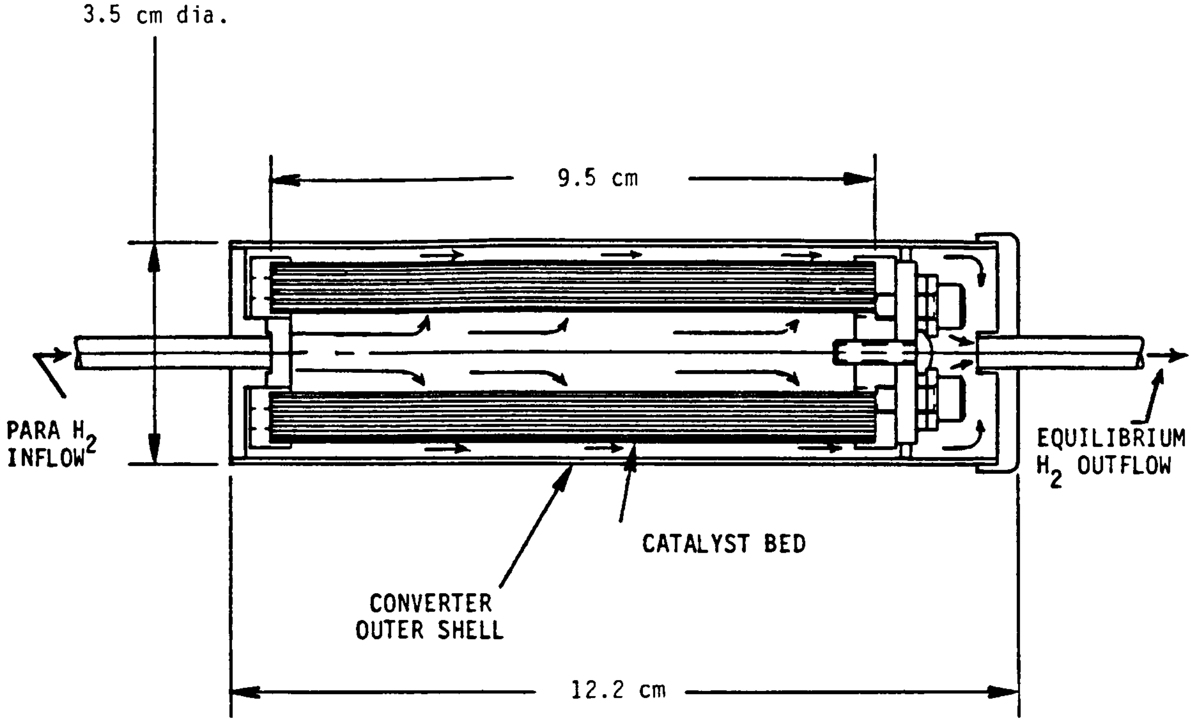


Figure 2-38. PARA-ORTHO CONVERTER DETAIL.

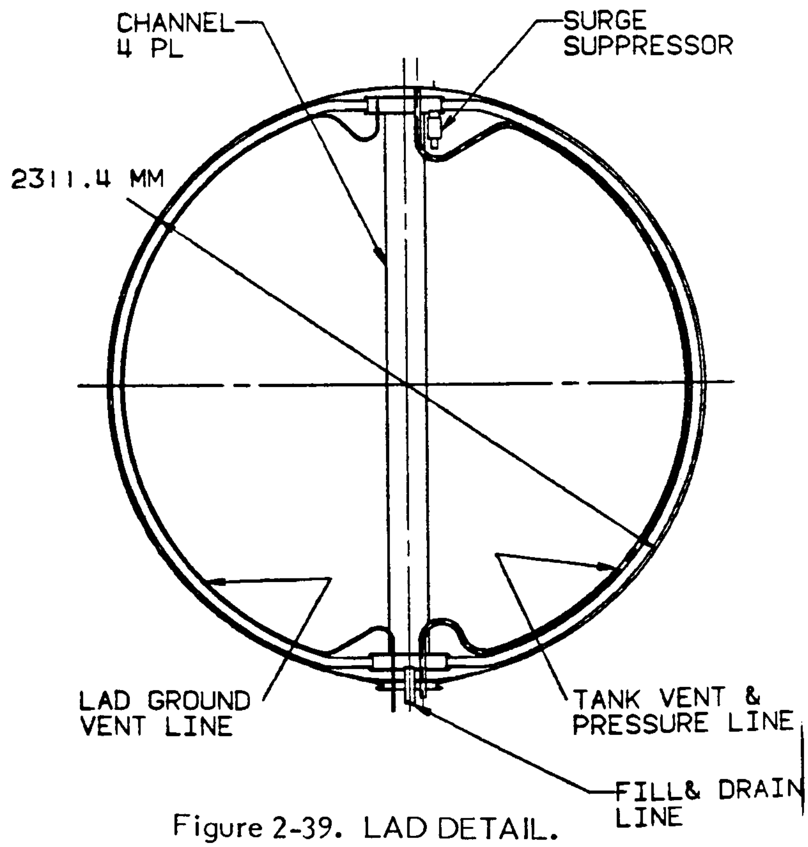


Figure 2-39. LAD DETAIL.

Table 2-XXII. PHASE I - SUPPLY DEWAR DESCRIPTION.

SUPPLY DEWAR - LH_2:

Modified OTTA - Vol = 6.46 m^3 (228 ft^3) flight weight PV and outer shell

TVS w/Joule-Thomson and PV wall HEX, Two Vapor Cooled Shields - 0.51 mm (.020 in) 6061 AL

MLI - 90 layers DAM MLI (ϵ = 0.035) / silk net

Pressure Vessel - Inner Vapor Cooled Shield (VCS)	15 layers
Inner VCS - Outer VCS	30 layers
Outer VCS - Outer Shell	45 layers

Para-Ortho Converter between Vapor Cooled Shields

Dual Stage Struts

PV - 231 cm ID x 0.89 mm thick, wt = 44 kg, 2219-T6 Al (91" ID x 0.035" wall, wt = 97 lbm)

OS - 266.4 cm OD x 3.58 mm wall, wt = 215 kg, 6061-T6 Al (104.9" OD x 0.141" wall, wt - 475 lbm

Silverized Teflon coating - α/ϵ = 0.2 (nominal outer shell temperature = 256 K (460°R)

Capillary Acquisition Device

RF Quantity Gaging System

Nominal Tank Pressure - 138 kPa (20 psi)

Nominal Tank Temperature - 21.3 K (38.4°R)

Thermal Performance
Heat leak - 0.88 W (3.02 BTU/hr), boiloff rate 0.0073 kg/hr (0.016 lbm/hr)

Total Dry Weight - 429 kg (945 lbm)

Total Wet Weight - 878 kg (1936 lbm)

PLUMBING:

Pressurization Line	1.27 cm x .71 mm wall x 203 cm (304 Cres) 0.5" x .028" wall x 80"
Fill/Drain Line	2.54 cm x .71 mm wall x 203 cm (304 Cres) 1.0" x 0.028" wall x 80"
Inner and Outer VCS	0.476 cm x .71 mm wall (6061 Al) 0.1875" x .028" wall

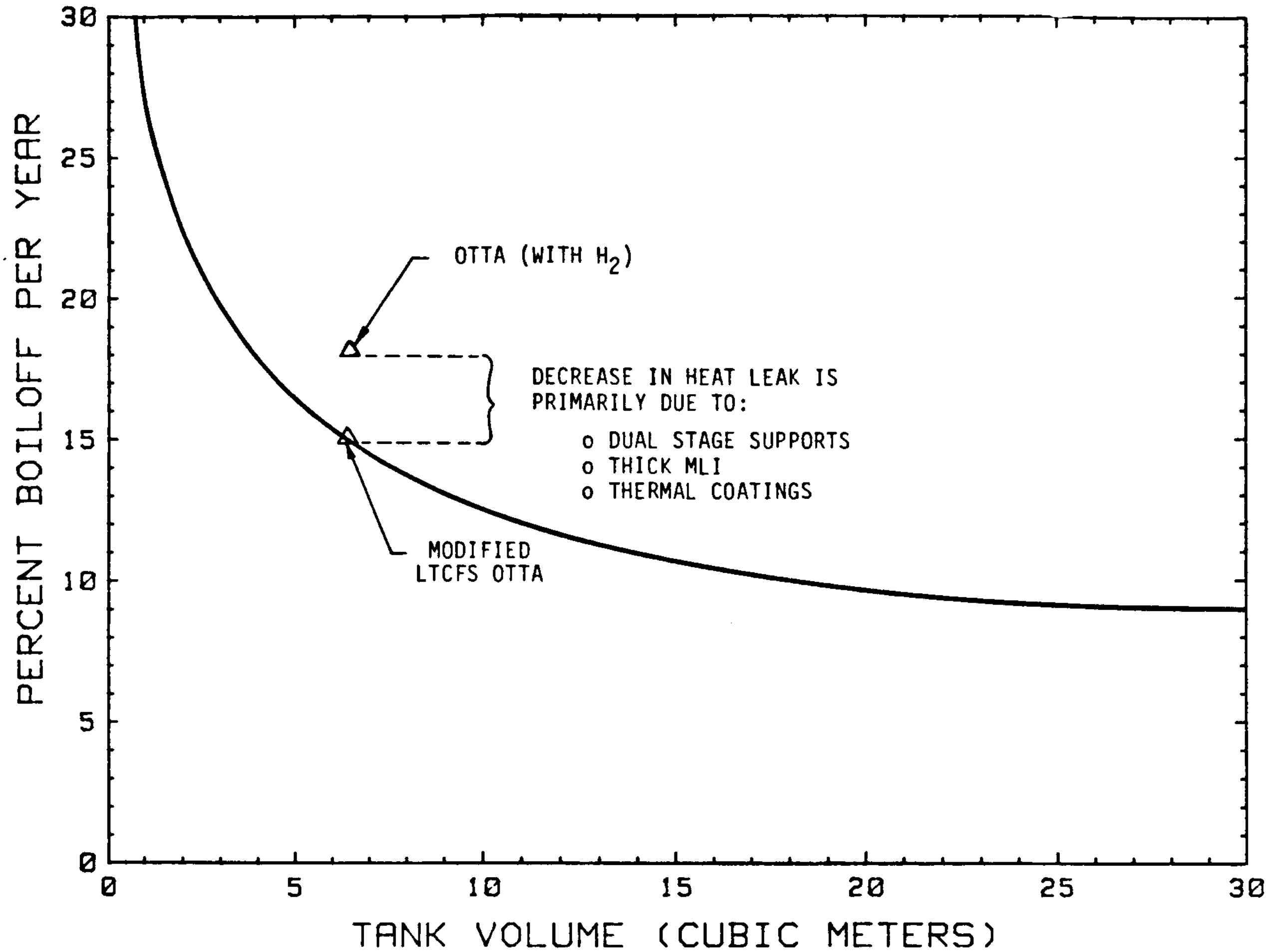


Figure 2-40. SUPPLY DEWAR THERMAL PERFORMANCE COMPARISON WITH OTTA.

A fluid schematic of the Phase I configuration is presented in Figure 2-41, indicating system flow lines, valving and instrumentation. Manual safety backup systems are provided in the event of system failure. For example, the vent and transfer lines contain a pressure relief valve in parallel with a burst disc should solenoid valving fail and create an overpressure situation. This dual failure tolerant system is needed to satisfy Space Station and STS flight safety requirements.

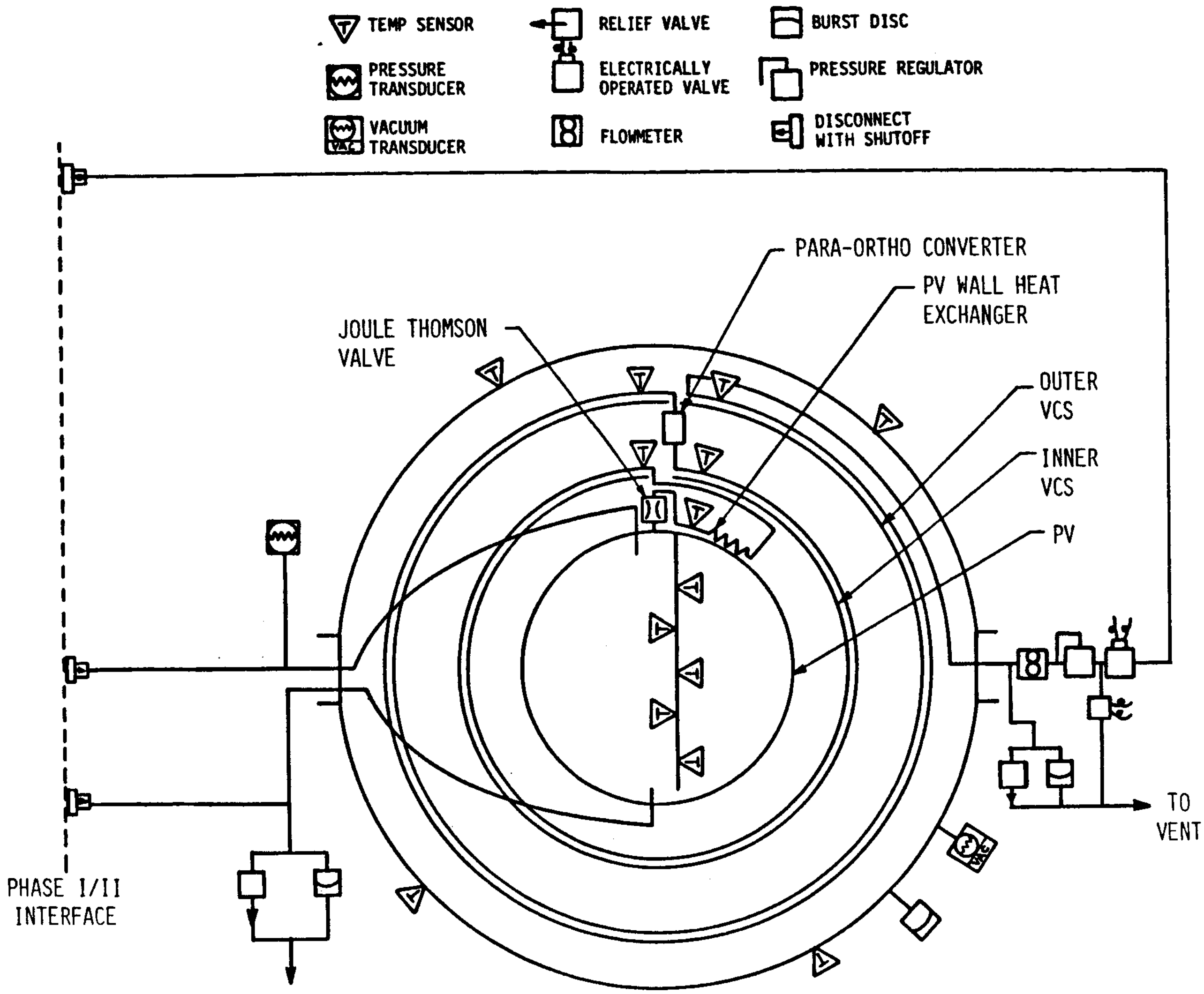


Figure 2-41. PHASE I SYSTEM SCHEMATIC.

A contingency dump system is required for STS flight safety. This system must dump the supply dewar cryogen in 250 seconds in the event of a RTLS Shuttle abort. A 0.31 m^3 (11 ft^3), 20.7 MPa (3000 psia) gaseous helium bottle is provided on the Phase I hardware for dump pressurization. This pressurization bottle is depicted in the Phase I configuration drawings, Figures 2-29, 2-30, 2-31, and in the dump system schematic, shown in Figure 2-42. The dump system is baselined to interface with the existing Centaur Orbiter Mod Kit, which provides dump and vent lines from the Shuttle payload bay to the Shuttle exterior surface. Cancellation of the Shuttle Centaur program may make use of this hardware questionable.

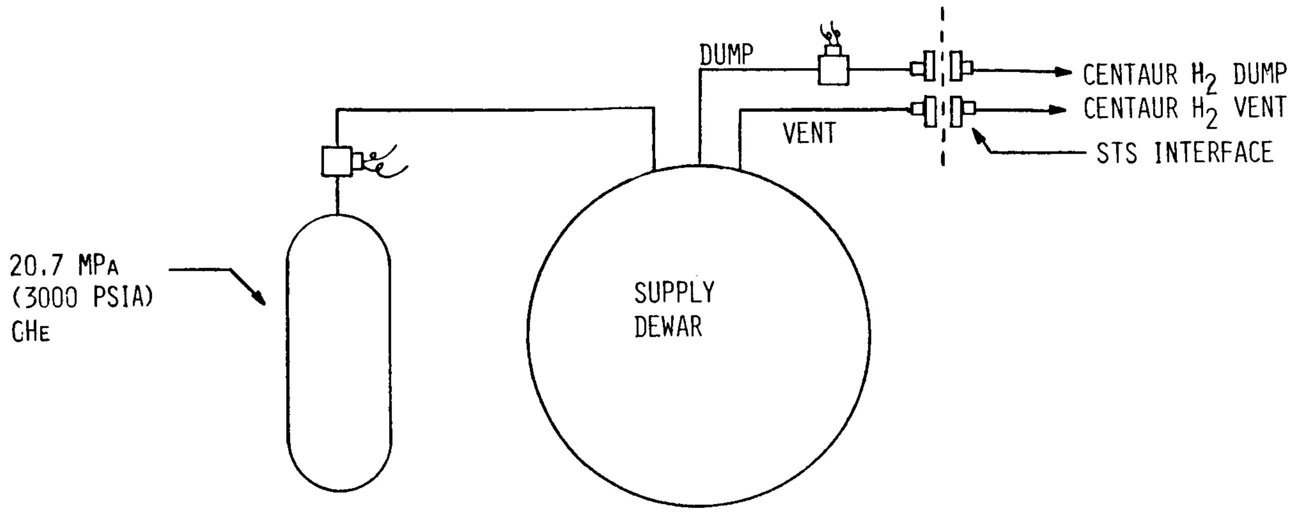


Figure 2-42. STS CONTINGENCY DUMP SCHEMATIC.

The Phase I instrumentation list is presented in Table 2-XXIII. Sensors compatible with the Space Station data acquisition system will be utilized whenever possible to minimize the amount of signal conditioning hardware required. All sensors in the experiment will have backups, since the length of the experiment increases the probability of sensor failure. All measurements are to be measured at a sampling rate of ten times per hour. Phase I data will be downlinked to a ground station once a week. Utilization of the Telescience system, described earlier, will allow real-time data to be accessed. The Space Station resource requirements for Phase I are summarized in Table 2-XXIV.

During STS launch, the Phase I hardware is located in the aft end of the payload bay, as shown in Figure 2-43. The payload center of gravity is located at STS station number 1175. This location meets the STS center of gravity constraints outlined in Reference 3 and also allows access to payload bay deployable keel and bridge fittings for experiment mounting. Access to the Centaur Mod Kit vent and dump interfaces is also made possible by this aft location.

Table 2-XXIII. PHASE I - INSTRUMENTATION LIST.

Cryogen Temperature (6)
PV Wall Temperature (4)
J-T Valve Exit Temperature
Pressure Vessel Heat Exchanger Exit Temperature
Inner and Outer VCS Exit Temperatures (2)
Outer Shell Temperature (4)
Tank Pressure
TVS Flowrate
P-O Converter Inlet and Outlet Temperature (2)
Tank Cryogen Quantity

Table 2-XXIV. PHASE I - RESOURCE REQUIREMENTS.

Electrical Power - 100 watts	
Crew Manpower Requirements:	
Deployment/Setup EVA	12 manhours
Deployment/Setup IVA	24 manhours
Data Downlink/Status Check IVA	1 manhour/week
Data Acquisition Interface:	
20 Temp. Transducers	Range 11 to 333 K (20 to 600°R)
Vacuum Transducer	Range 10^{-4} to 10^{-9} Torr
Pressure Transducer	Range 0 to 345 kPa (0 to 50 psia)
Mass Flowmeter	Range 0 to 0.045 kg/hr (0 to 0.1 lbm/hr)
Quantity Gaging System	

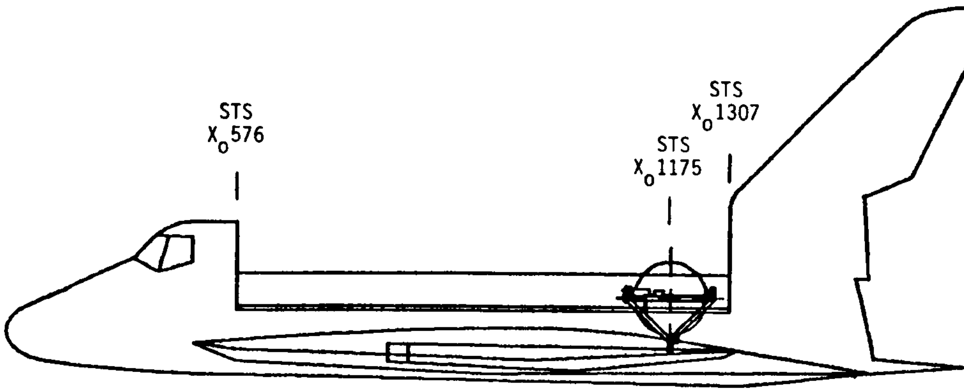


Figure 2-43. PHASE I PAYLOAD BAY LOCATION.

The experiment will be deployed from the Shuttle payload bay to the Space Station structure by using the Shuttle and/or Space Station Remote Manipulator System (RMS). A standard grapple fixture, shown in Figures 2-29 and 2-30, will be located on the support structure for use by the RMS. The experiment will be mounted to the Space Station structure as shown in Figure 2-44. Adjustable tripods will be attached to the Space Station structure. The RMS will position the Phase I module over the tripods. The apex of each tripod will then be attached to the trunnion pins that were previously utilized for Shuttle payload bay mounting. A detailed view of a tripod leg is presented in Figure 2-45. Each leg has a fitting on one end for attachment to the Space Station structure and a trunnion pin attach fitting on the other end. Tripod length may be adjusted using a ratchet mechanism for large adjustments and a turnbuckle for fine adjustments.

After the hardware has been mounted, an EVA will be performed to connect the Space Station power and data interfaces. Operational checkout of the experiment will then be performed to verify the experiment is functioning properly. The hardware will be allowed to reach a quasi-steady state condition (approximately 2-3 months after deployment) and then long-term performance will be measured. The Phase I time span of two years will allow evaluation of performance degradation due to the orbital environment. Analytical models created during the experiment design effort will be correlated to test data. This correlation effort will provide a benchmark for future orbital cryogenic storage systems.

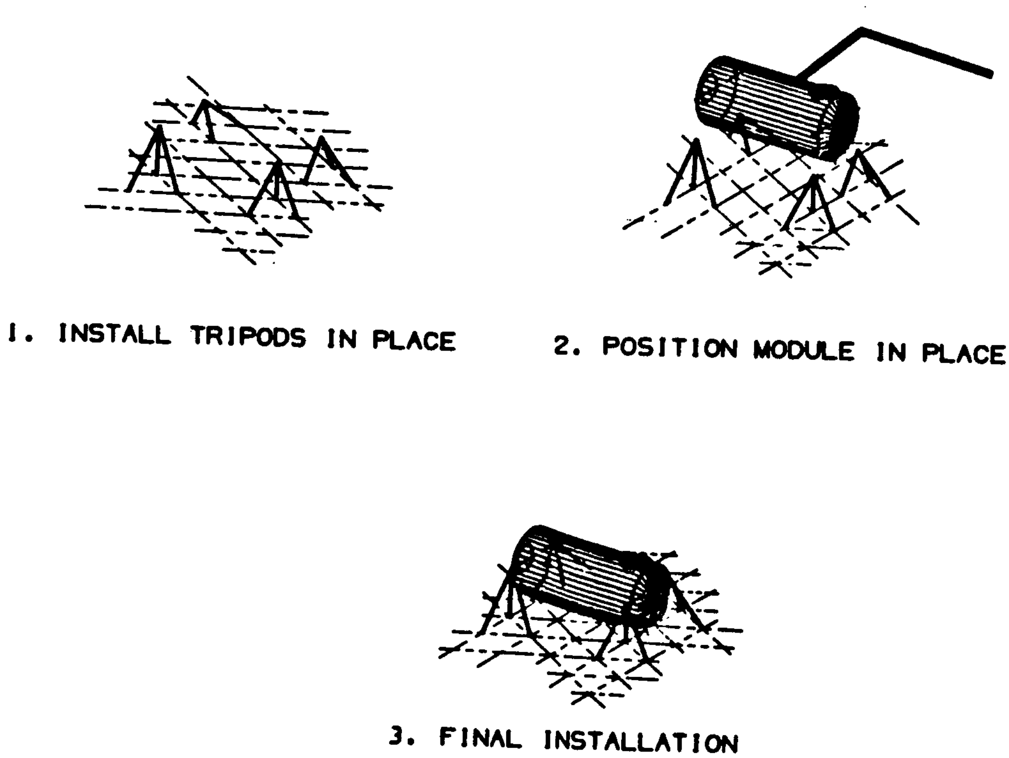


Figure 2-44. SPACE STATION DEPLOYMENT AND MOUNTING.

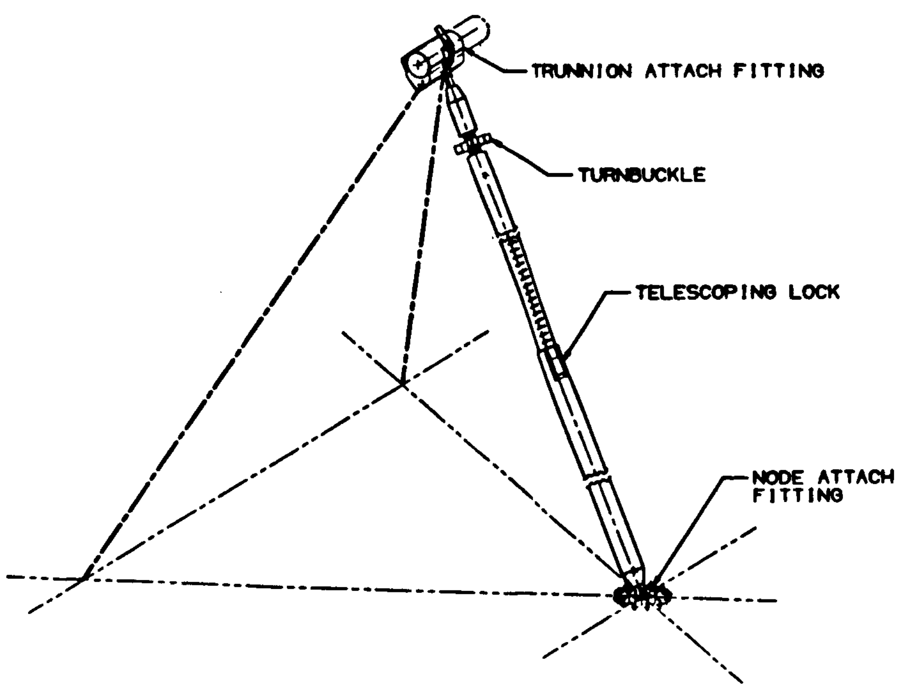


Figure 2-45. TRIPOD LEG DETAIL.

2.4.2.2 Phase II Description. Phase II of the experiment is designed to demonstrate and evaluate low-g fluid transfer technologies. In addition, thermal performance of the soft outer shell receiver tank will be evaluated. An isometric view of the Phase II experiment configuration is presented in Figure 2-46. The Phase I hardware will be reconfigured on-orbit by the addition of the Phase II module containing the receiver tank and pressurization system. Fluid transfer operations will then be performed to evaluate the hardware and techniques necessary to achieve low-g fluid transfer. The receiver tank in the Phase II module will be flown up to Space Station empty, eliminating many flight safety issues and the requirement for ground purge of the soft outer shell receiver tank. This reduces the cost and complexity of the Phase II module. Isometric and three-view drawings of the Phase II module are shown in Figures 2-46 through 2-50.

The Phase II receiver tank is a 1.27 m^3 (45 ft^3) modified ELMS soft outer shell tank. A transfer line wrapped in MLI connects the supply and receiver tanks for fluid transfer. The MLI is not shown in the drawings for purposes of clarity. A line providing gas pressurant from the Phase II pressurization system is interfaced to the supply dewar. Supply dewar vent gas is routed to a boiloff collection system on the Phase II module, where the gas is stored and pressurized utilizing a metal hydride compressor to provide gas pressurant for fluid transfer operations. The pressurant is stored in a 3.4 MPa (500 psia) spherical aluminum pressure vessel.

Electrical power is provided from the Phase I module via the electrical interface panel. Data from Phase II instrumentation is also routed through this panel to the Phase I DACS. Cooling of the metal hydride compressor is provided by the Space Station thermal bus. An interface panel for the thermal bus is located on the lower portion of the support structure. The Phase II support structure is similar in design to Phase I, with STS trunnion mounting pins that are utilized both as payload bay and Space Station mounts. An RMS grapple fixture is fixed to the structure for experiment deployment and retrieval.

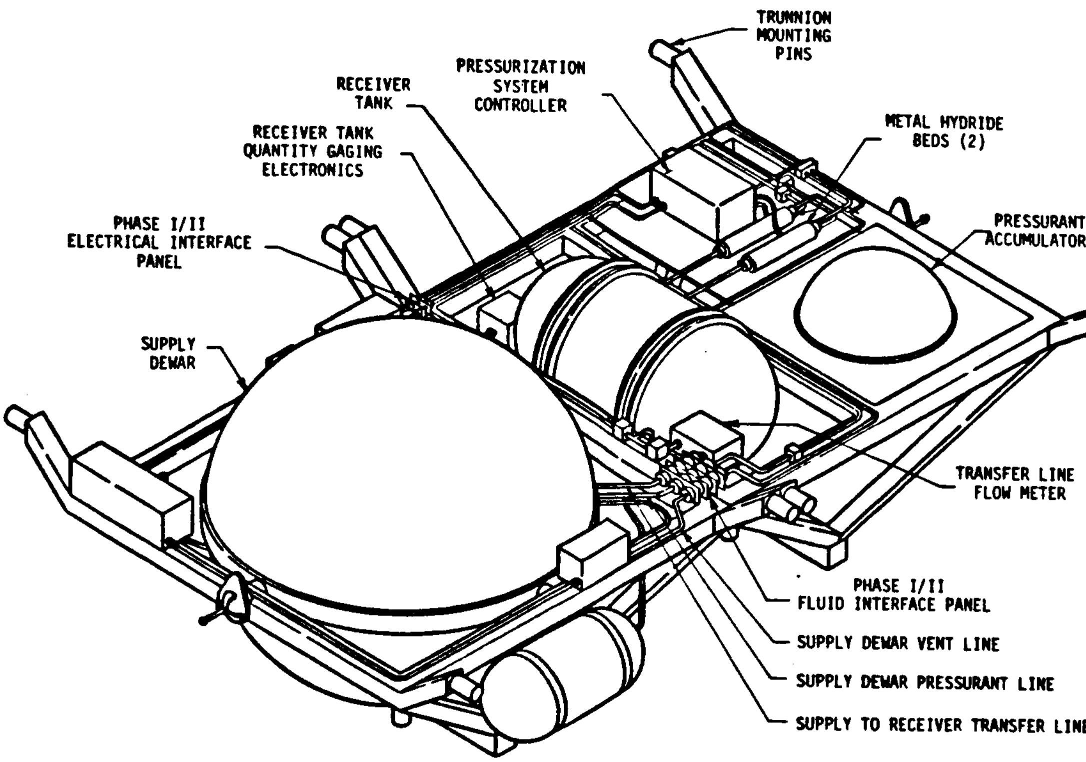


Figure 2-46. PHASE II CONFIGURATION - ISOMETRIC VIEW.

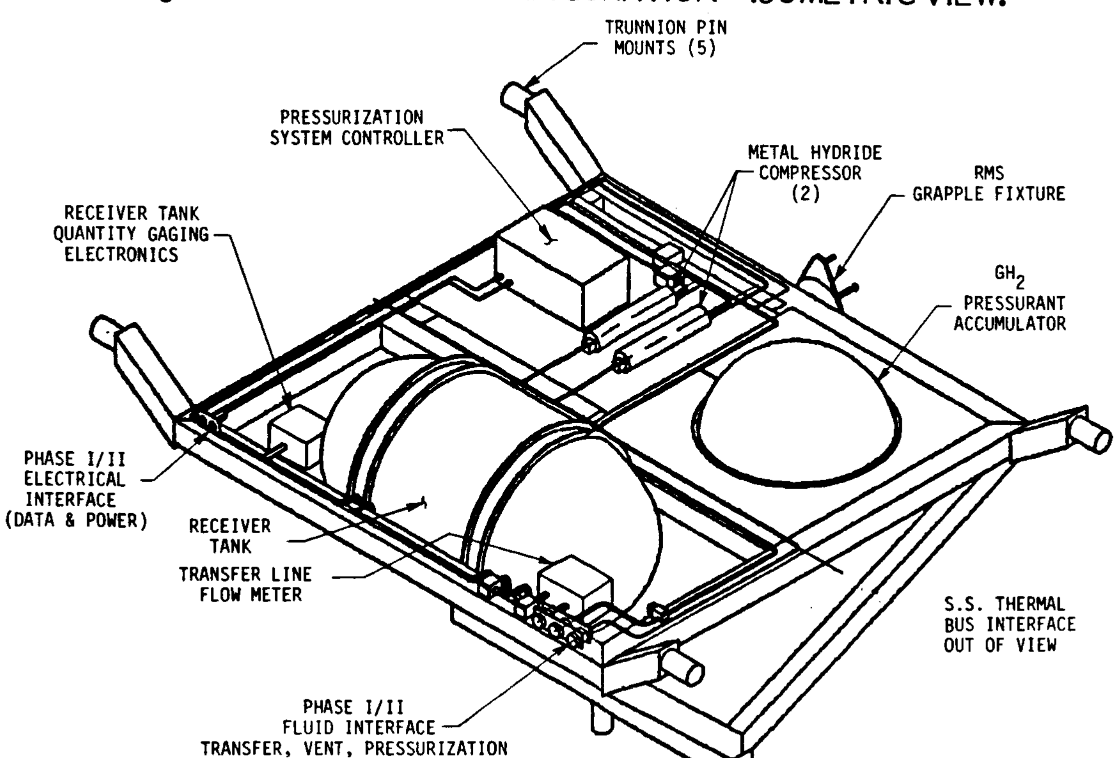


Figure 2-47. PHASE II MODULE - ISOMETRIC VIEW.

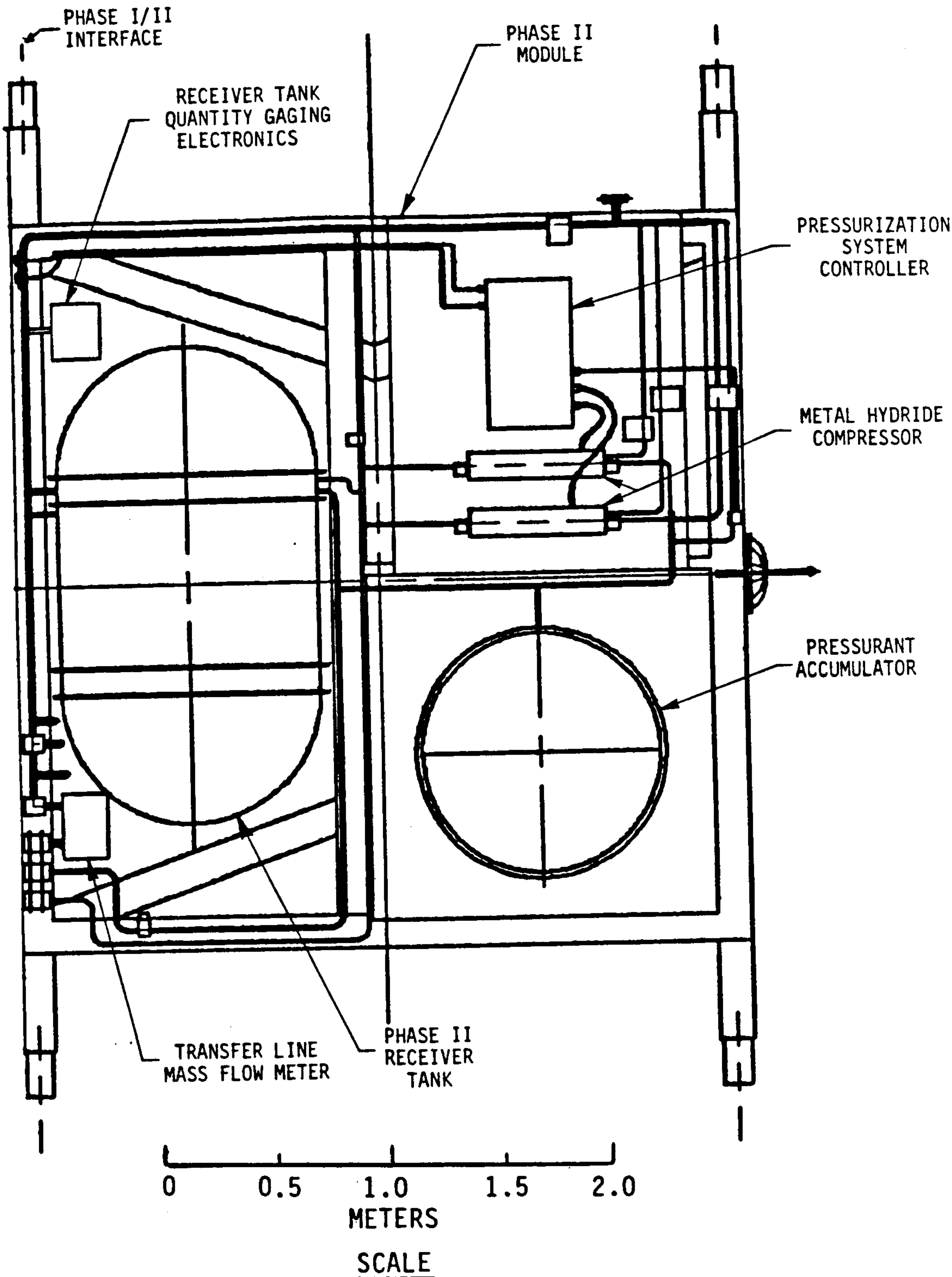


Figure 2-48. PHASE II MODULE - TOP VIEW.

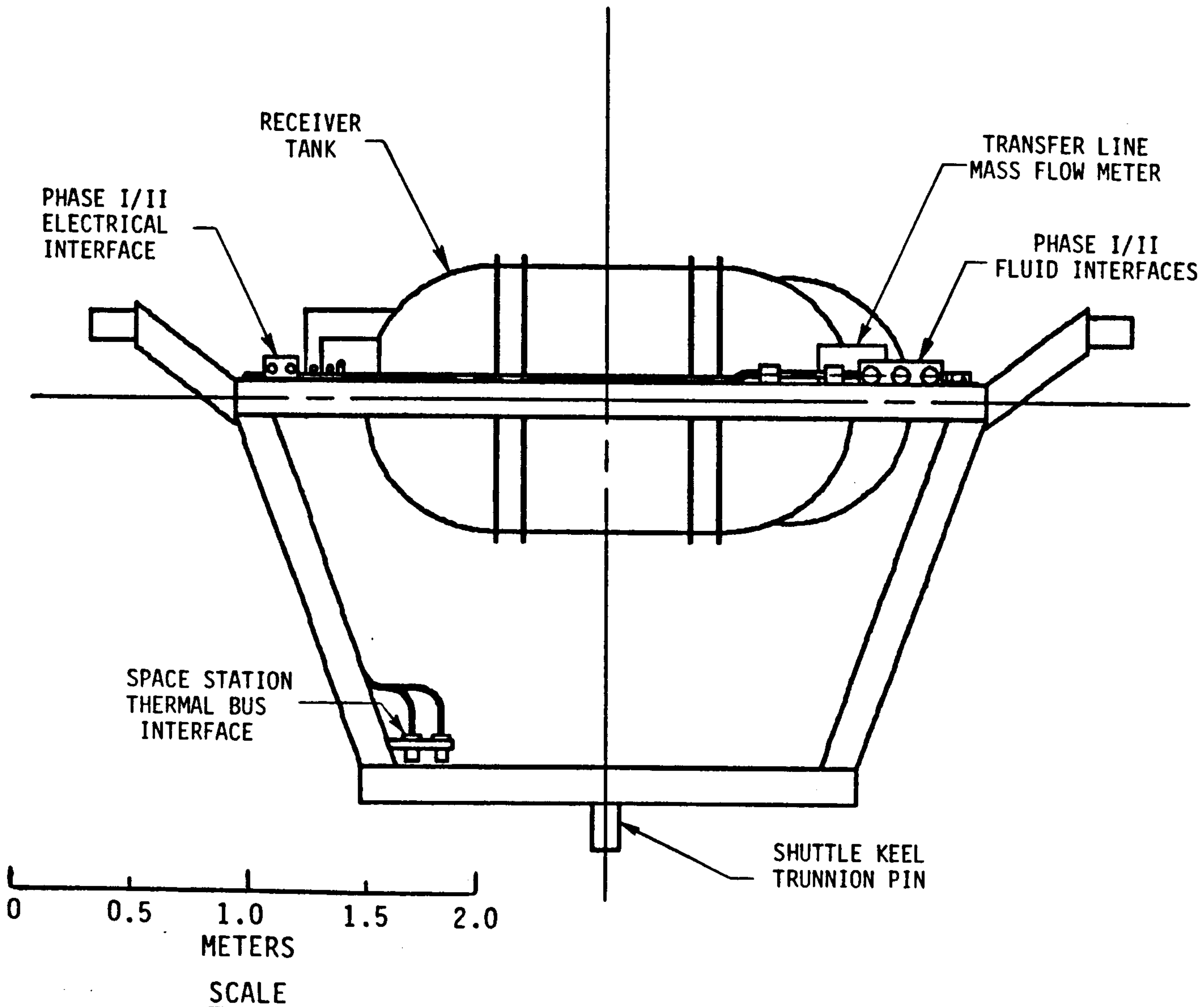


Figure 2-49. PHASE II MODULE - FRONT VIEW.

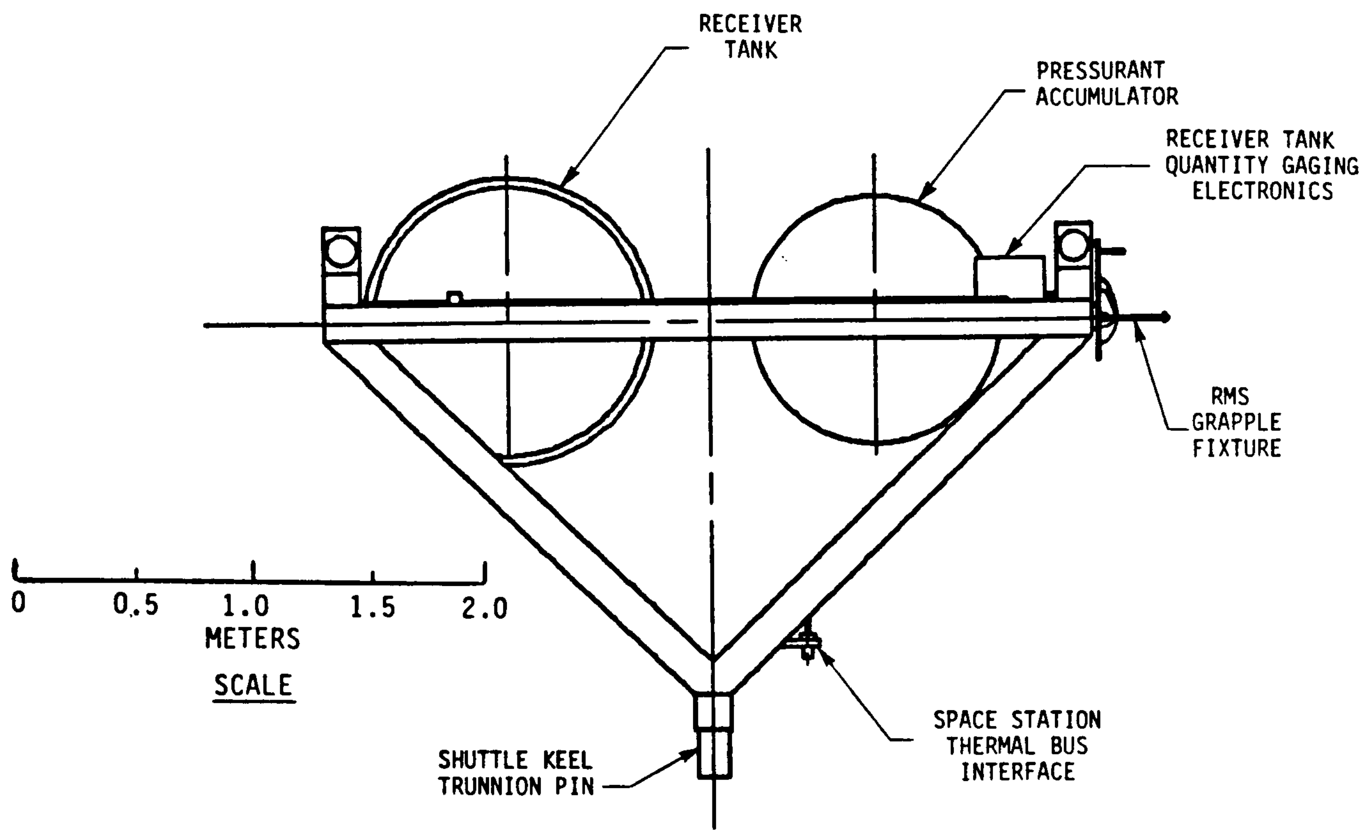


Figure 2-50. PHASE II MODULE - SIDE VIEW.

The receiver tank, as previously described in Section 2.4.1, is a 1.27 m^3 (45 ft^3) modified ELMS tank. The tank utilizes an RF quantity gaging system to measure cryogen mass, and a LAD for fluid acquisition during transfer operations. Both of these systems are similar in design to those present in the supply dewar. The receiver tank utilizes a TVS similar to the supply dewar for tank venting, but with no vapor cooled shields. A no-VCS system was chosen because high thermal performance is not necessary in the receiver tank. The tank insulation system consists of 60 layers of double-aluminized mylar with an external 1.7 mm (0.067 inch) thick micro-metroid shield. The micro-meteroid shield is coated with a silverized teflon laminate to reduce tank heat leak. The receiver design parameters are summarized in Table 2-XXV.

Table 2-XXV. RECEIVER TANK DESCRIPTION.

<u>RECEIVER TANK:</u>
Modified ELMS, Flight Weight PV Volume = 1.27 m^3 (45 ft^3)
No Vacuum Jacket, 1.7 mm (0.0670") 6061 Al Micrometeroid Shield (shield and MLI constitutes micrometeroid protection system)
TVS w/Joule-Thomson Valve and PV Wall HEX (no VCS)
Axial, Radial, and Tangential Spray Nozzle
MLI - 60 Layers DAM MLI / Silk Net $\epsilon = 0.035$
Strut Suspension System sized for empty PV Flight Loads, A/L approximately 0.02" = 0.051 cm (0.00167 ft)
Micrometeroid Shield - 115 cm ID x 1.7 mm (45.38" ID x 0.067") 6061 Al
Capillary Acquisition Device
Quantity Gaging System
Pressurization and Fill Lines - 1.27 cm dia x 0.71 mm wall x 127 cm (0.5" dia x 0.028" wall x 50") 304 Cres
TVS Line - 0.476 cm dia x .71 mm wall (0.1875" x 0.028" wall)
Total Line A/L - 4.48×10^{-3} cm (1.47×10^{-4} ft)
Heat Leak - 1.32 W (4.49 BTU/hr)
Boiloff Rate - 0.0104 kg/hr (0.023 lbm/hr)
Transfer line - 1.27 cm x 0.71 mm wall (1/2" x 0.028" wall), 30 layers DAM/silk MLI = 0.035
Hydride Boiloff Collection Pressurization System
Receiver Tank Mass - 129 kg (284 lbm)

Details of the metal hydride compressor are shown in Figure 2-51. Film heaters and coolant tubes are wrapped around the exterior of the vessel to provide heating and cooling as necessary. The coolant tubes are interfaced with the Space Station thermal bus and will be MLI wrapped. The entire compressor assembly will also be wrapped in MLI to reduce heating and cooling requirements. Fittings on each end of the compressor allow flow of GH_2 to and from the compressor. These fittings contain filters to prevent metal hydride dust from exiting the compressors.

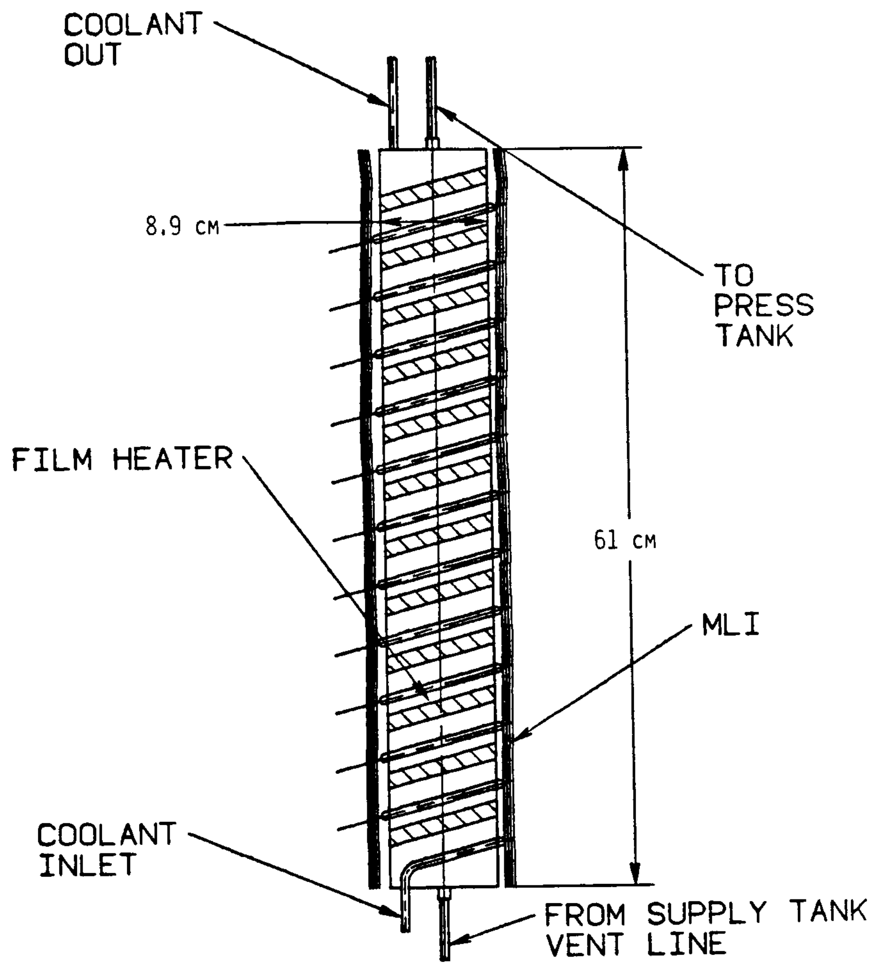


Figure 2-51. HYDRIDE COMPRESSOR DETAIL.

Fluid schematics of the Phase II configuration are presented in Figures 2-52 through 2-54. Each schematic depicts a different mode of system operation. Figure 2-52 depicts the standby mode. In this mode, the receiver tank is empty, and supply tank boiloff is being collected for pressurization.

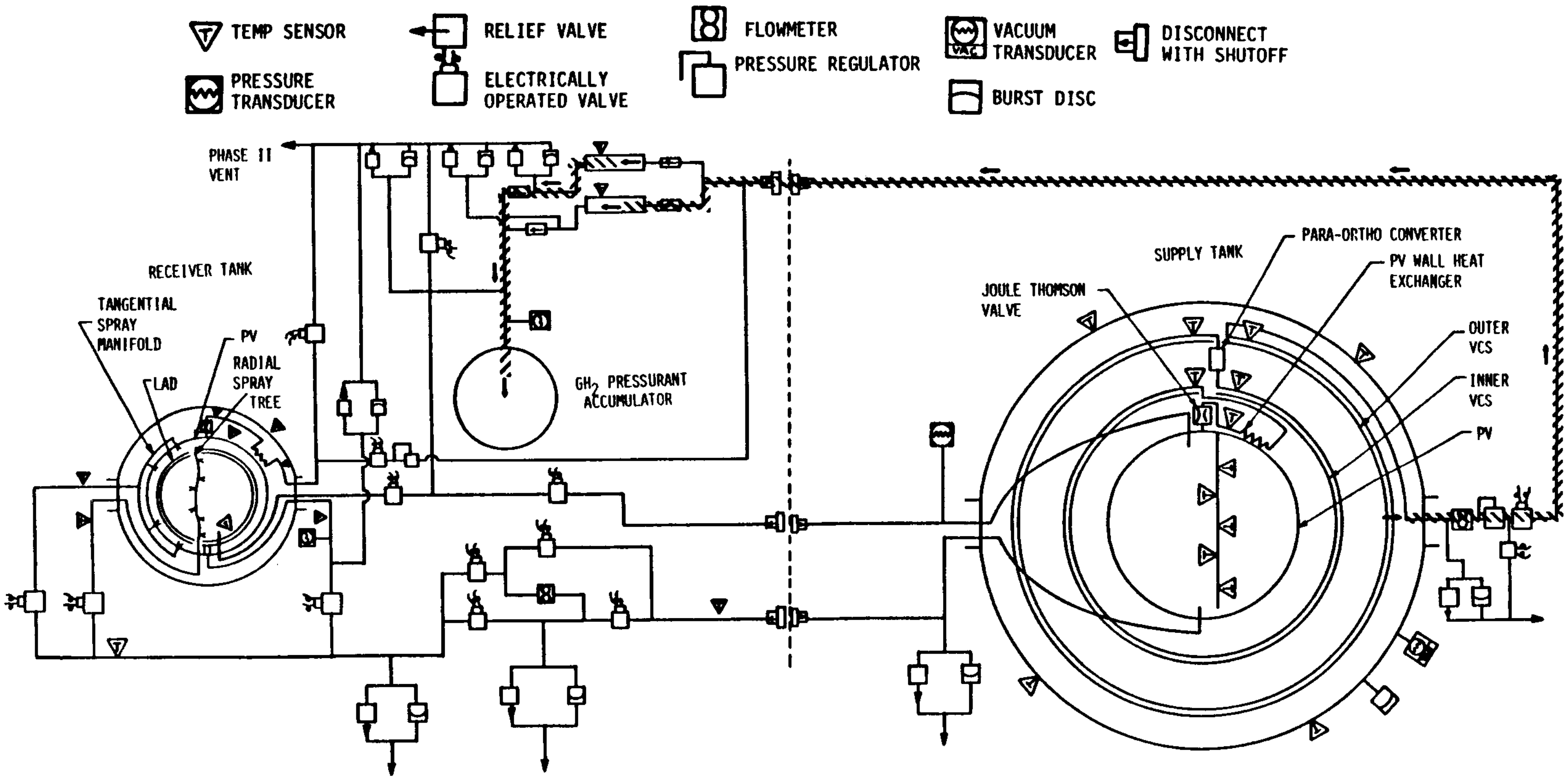


Figure 2-52. PHASE II SCHEMATIC - STANDBY OPERATION.

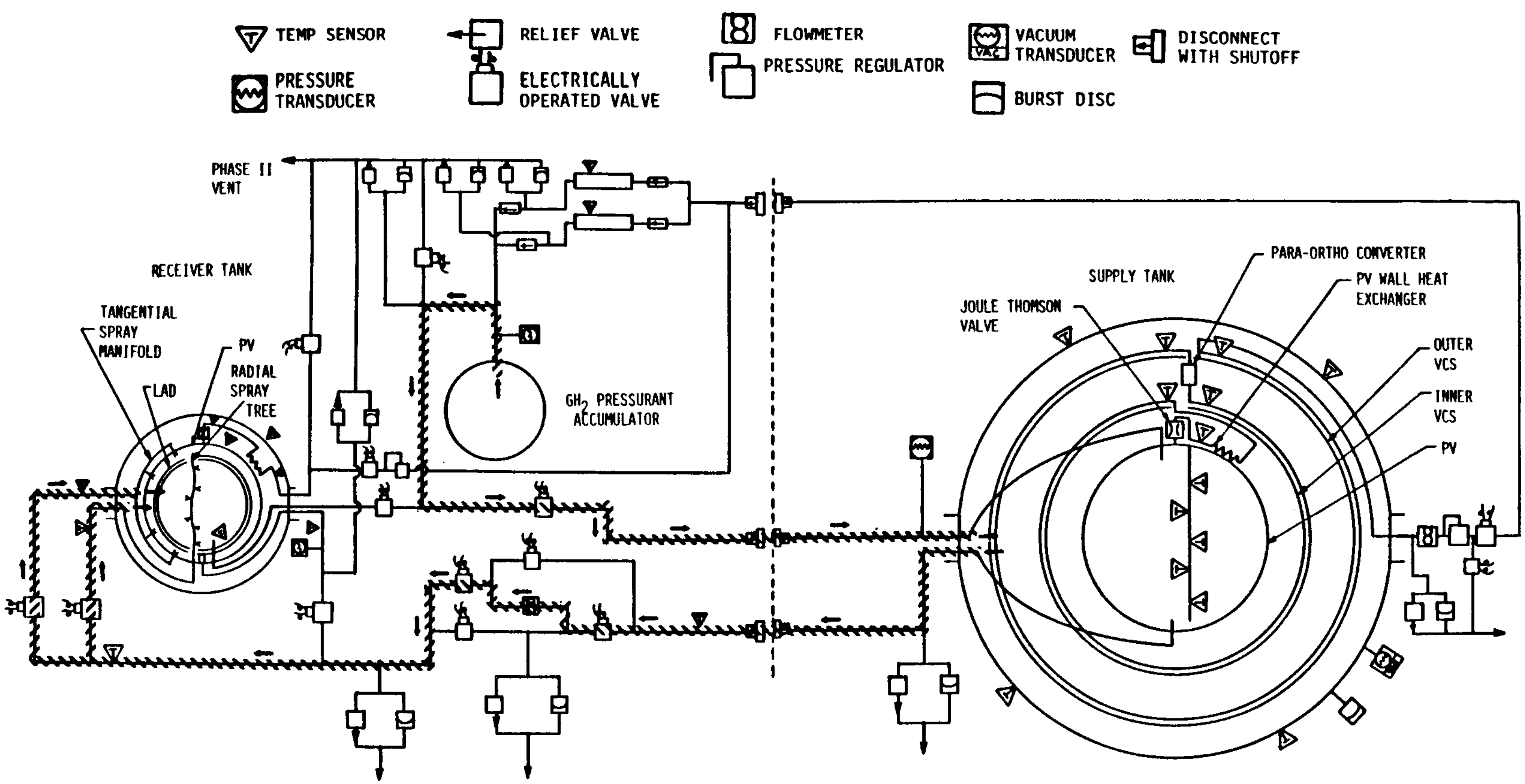


Figure 2-53. PHASE II SCHEMATIC - RECEIVER TANK COOLDOWN.

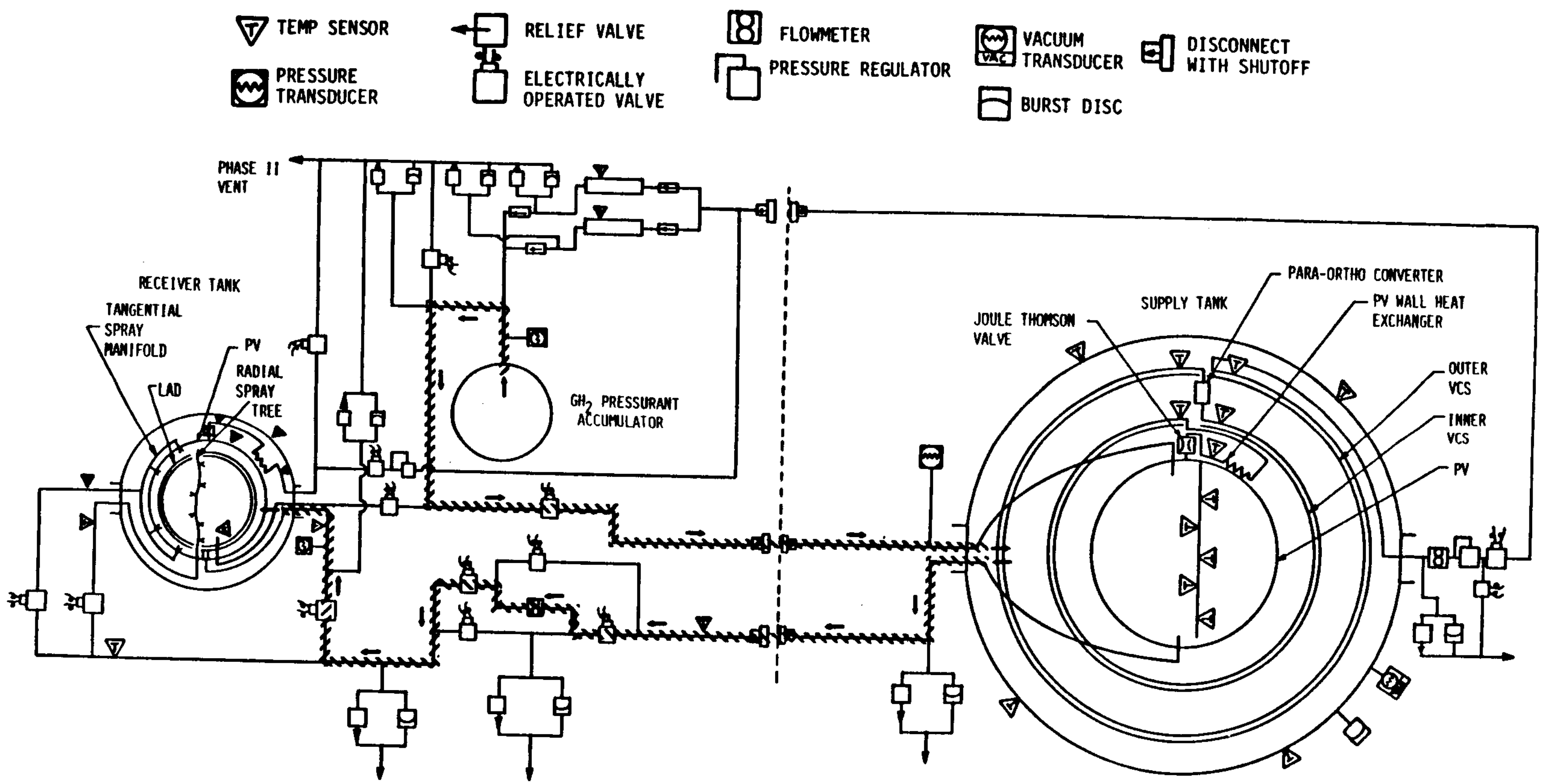


Figure 2-54. PHASE II SCHEMATIC - FLUID TRANSFER.

This mode of operation must be performed for two weeks prior to a fluid transfer operation in order to collect adequate pressurant gas. The hatched line with arrows depicts the fluid flow path. Boiloff gases exit the supply tank TVS, passing through the TVS flowmeter and valving and are absorbed by the metal hydride bed as it is cooled. The second hydride bed, which is charged with H_2, is being heated, expelling the H_2 into the pressurant accumulator. This process continues until the pressurant accumulator reaches 3.45 Mpa (500 psia) at which point there is adequate pressurant to perform a transfer operation.

Prior to a transfer operation, transfer line and receiver tank cooldown will be performed. This mode of operation is depicted in Figure 2-53. Pressurant flows from the accumulator into the supply tank, expelling liquid into the transfer line. Initially, the transfer line is warm, and boiling occurs within the line, injecting vapor into the receiver tank. Once the transfer line has been cooled, chilldown of the receiver tank begins. As discussed in Section 2.4.1, tank chilldown consists of repeated cycles consisting of charging the tank with cryogen, holding the tank in a no-vent state while the cryogen is superheating, then venting the superheated vapor. Figure 2-53 depicts the cooldown flow during the charge cycle. The receiver tank has two injection systems for the cooldown fluid; a radial spray tree that sprays liquid from the center of the tank radially outward, and a tangential spray manifold that injects fluid tangentially along the tank wall. One or both of these spray systems may be used during tank cooldown and are controlled using separate valves. The cooldown schematic depicts both spray systems being used. After the fluid is injected and superheated, the vent valve is opened and the fluid is vented out the Phase II vent. This vented fluid can be collected and utilized on Space Station or vented overboard. If the fluid is vented overboard, a converging-diverging nozzle will be utilized to impart Space Station escape velocity to the vapor. A resistojet can also be utilized to impart added velocity if necessary.

After tank cooldown has occurred, fluid will be transferred from the supply tank to the receiver tank. This process is depicted in Figure 2-54. Pressurant gas flows from the accumulator to the supply tank, expelling fluid through the transfer line. The fluid passes through the mass flowmeter and enters the receiver tank through the fill line. It should be noted that the system is configured to flow cryogen from the receiver tank back to the supply dewar by pressurizing the receiver tank and backflowing through the transfer line. Valves and lines have been designed such that liquid will flow through the mass flowmeter in the same direction, regardless of direction of fluid transfer.

The Phase II instrumentation list is shown in Table 2-XXVI. This instrumentation is in addition to those listed for Phase I. During standby mode, data will be sampled at a rate of ten times per hour, and downlinked weekly, as in Phase I. This data sampling rate will also be utilized during receiver tank thermal performance testing. During cooldown and fluid transfer operations, data will be sampled at a frequency of 1 Hz, due to the highly transient nature of these operations. Fluid transfer operation data are to be downlinked at the termination of each transfer. Transfer operations will occur approximately every two weeks. The Phase II resource and interface requirements are summarized in Table 2-XXVII.

Table 2-XXVI. PHASE II - INSTRUMENTATION LIST.

Receiver PV Temperature
Receiver Tank J-T Exit Temperature
Receiver Tank PV HEX Exit Temperature
Transfer Line Temperature (5)
Transfer Line Flowrate
Receiver Tank Pressure
Receiver Tank Quantity
Hydride Compressor Temperature (2)
Hydride Compressor Pressure (2)
Accumulator Pressure

The Phase II module will be integrated with the Shuttle in the same fashion as Phase I, utilizing trunnion pin mounts. Upon reaching Space Station, the Phase II module will be deployed on the RMS and mounted to the Space Station in the same manner as the Phase I module. Utilizing an EVA operation, the Phase II module will be structurally attached to the Phase I hardware and the Phase I/II fluid and electrical interfaces will be attached. The Phase II Space Station thermal bus interface will then be attached. Hardware operation will then be checked out and verified prior to beginning experimentation.

Table 2-XXVII. PHASE II - RESOURCE AND INTERFACE REQUIREMENTS.

Electrical Power - 600 watts during fluid transfer operations, 100 watts idle	
Cooling - 15 watts via Space Station Thermal Bus	
Crew Manpower Requirements:	
Deployment/Setup EVA	8 manhours
Deployment/Setup IVA	24 manhours
Transfer Operation IVA	4 manhours per transfer
Data Downlink/Status Check IVA	1 manhour/week
Additional Data Acquisition Interfaces:	
12 Temperature Transducers	Range 11 to 333 K (20 to 600°R)
Mass Flowmeter	Range 0 to 182 kg/hr (0 to 400 lbm/hr)
Receiver Tank Pressure Transducer	Range 0 to 345 kPa (0 to 50 psia)
Accumulator Pressure Transducer	Range 0 to 3.45 MPa (0 to 500 psia)
Receiver Tank Quantity Gaging System	
Data Sample Rate - 10 per hour standby, 1 Hz during fluid transfer	

After experiment operation is checked out, the experiment will then be on standby mode for two weeks to collect boiloff gases for pressurization. Fluid will then be transferred to the receiver tank and receiver tank thermal performance will be measured for a 90-day period. Following receiver tank thermal performance testing, ten fluid transfer operations will be performed, one every two weeks. The operations will be performed utilizing varying cooldown and fill flowrates and cooldown methods, in order to determine optimal fill processes. The receiver tank fluid will be backflowed into the supply tank after each transfer operation in order to conserve cryogen and to demonstrate low-g refill of a partially full tank. After the last transfer operation to the receiver tank is performed, receiver thermal performance will again be measured for a 90-day period to measure any degradation of thermal performance. During this last receiver tank thermal performance test, the supply tank will have a 75% ullage. The effect this high ullage has on the supply tank LAD and thermal performance will be investigated during this period. At the end of the thermal performance test, the receiver tank fluid will be backflowed into the supply tank, terminating Phase II operations. Phase II testing will last approximately one year.

2.4.2.3 Phase III Description. Phase III of the experiment will demonstrate active refrigeration technologies. In Phase III, a refrigeration unit will be integrated with the Phase I supply dewar to reduce or eliminate net heat leak to the cryogen. Long lifetime, flight qualified refrigerators are the least developed of all technologies that are to be included in the LTCFSE, yet they also have the most technology development programs currently underway. These development programs encompass a wide variety of refrigerator types from closed-gas cycles, such as the Stirling and Brayton cycles, to gas absorption and magnetic refrigerators. Several of these refrigerators, most notably the Vuilleumier and several Stirling cycle machines, have demonstrated several thousands of hours of continuous operation, including one type flown on the DOD P-78-I Satellite in 1979 (Phillips Rhombic Drive Stirling). However, it is still unclear as to which particular unit will prove best suited for use on Phase III of the LTCFSE experiment. Thus, it was decided to design the Phase III hardware in a "generic" manner capable of interfacing with several types of refrigeration units with minimal changes. Any refrigeration unit requires four basic interfaces:

1. Electrical input power (or heat which may be electrically derived)
2. Coolant Interface
3. Waste Heat Transfer
4. Instrumentation and Control

The Phase III hardware contains these interfaces, allowing the design to be suitable for several types of refrigeration units.

An isometric view of the Phase III experiment configuration is presented in Figure 2-55. The Phase III module is attached to the side of the Phase I module and interfaced with the Phase I data and fluid systems. The Phase III Space Station interfaces for cooling and electrical power are shown on the front of the Phase III module. A top view of the Phase III configuration is shown in Figure 2-56. Top and side views of the Phase III module, defining major components and subsystems, are shown in Figure 2-57.

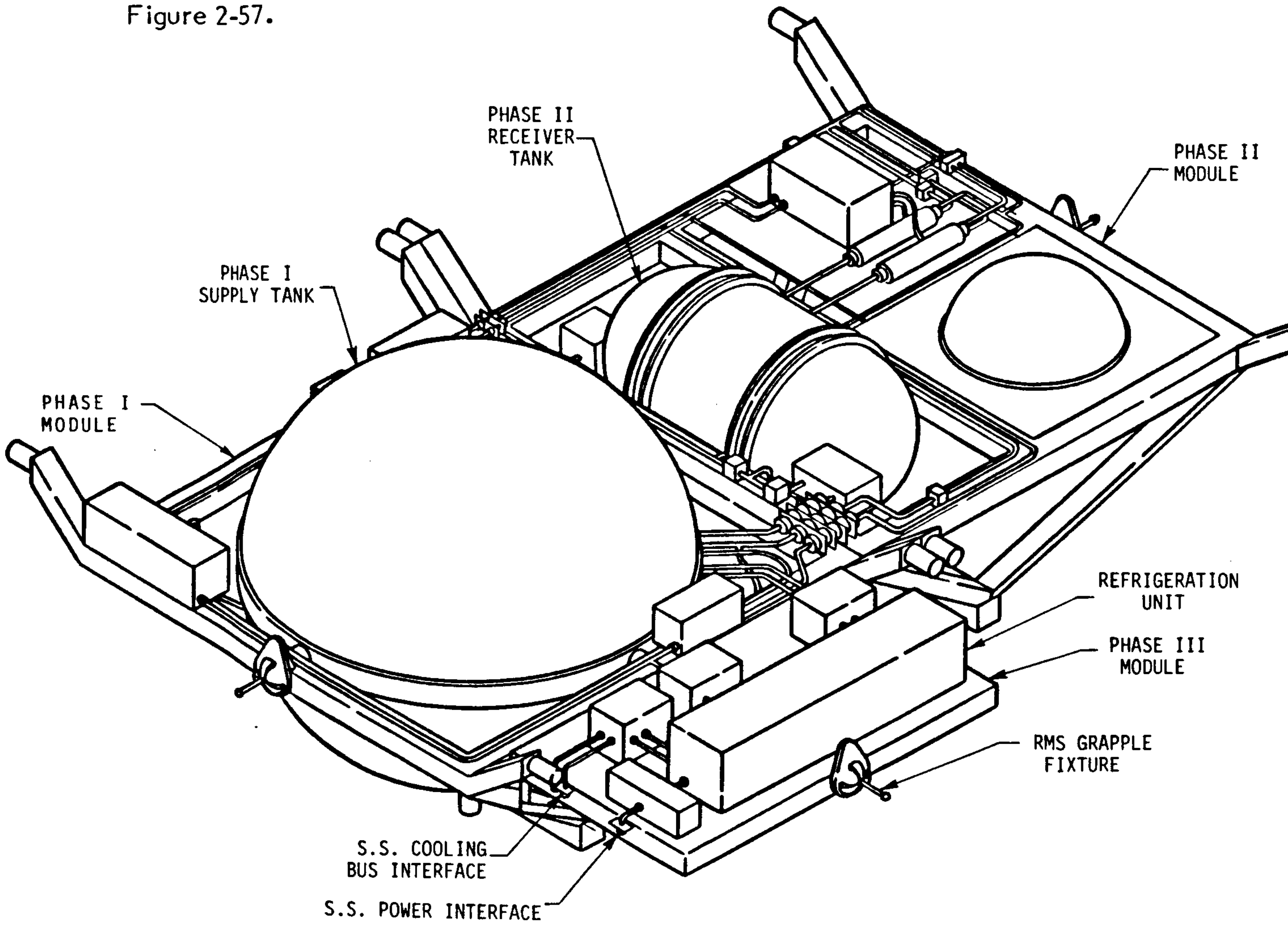


Figure 2-55. PHASE III EXPERIMENT CONFIGURATION - ISOMETRIC VIEW.

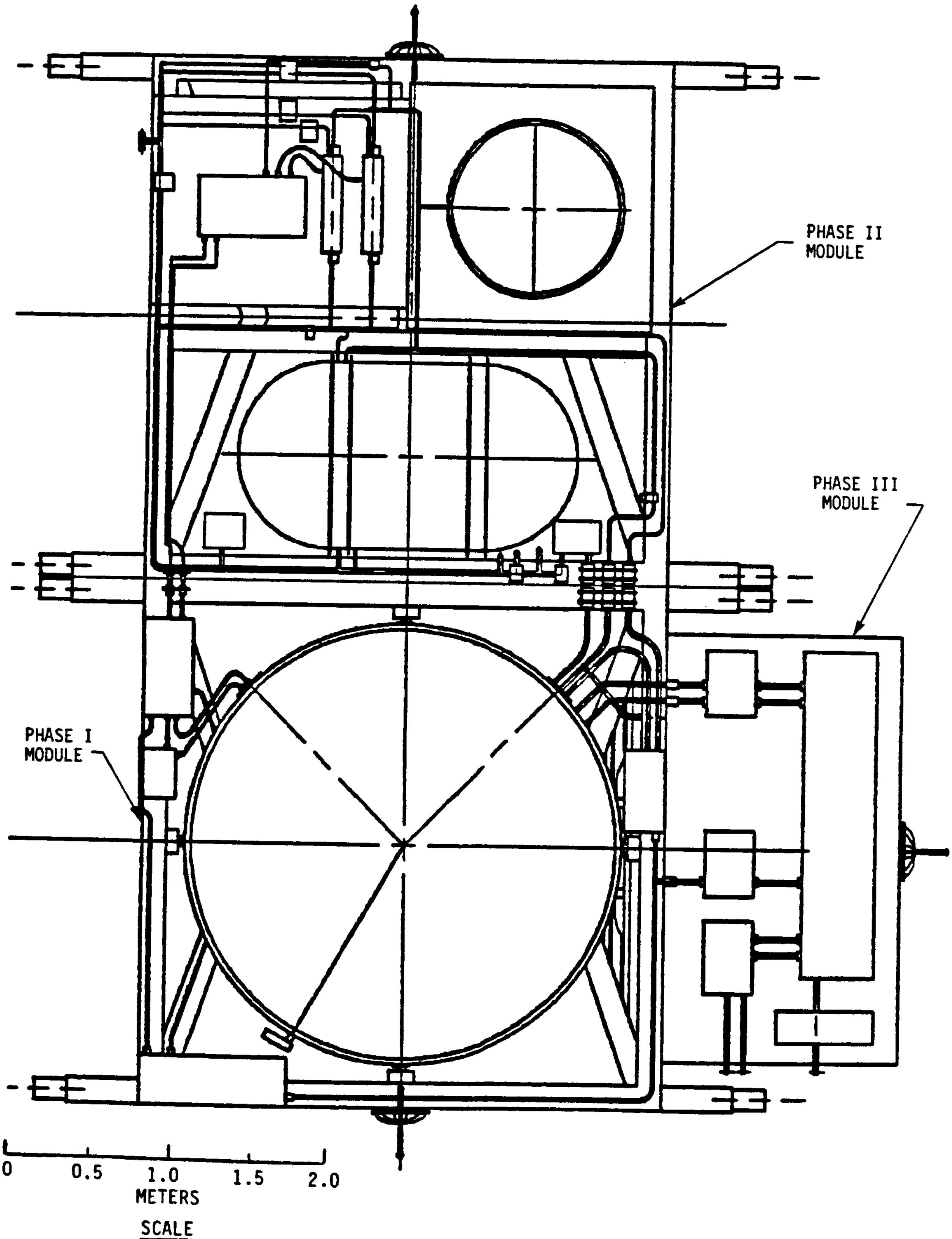


Figure 2-56. PHASE III EXPERIMENT CONFIGURATION - TOP VIEW.

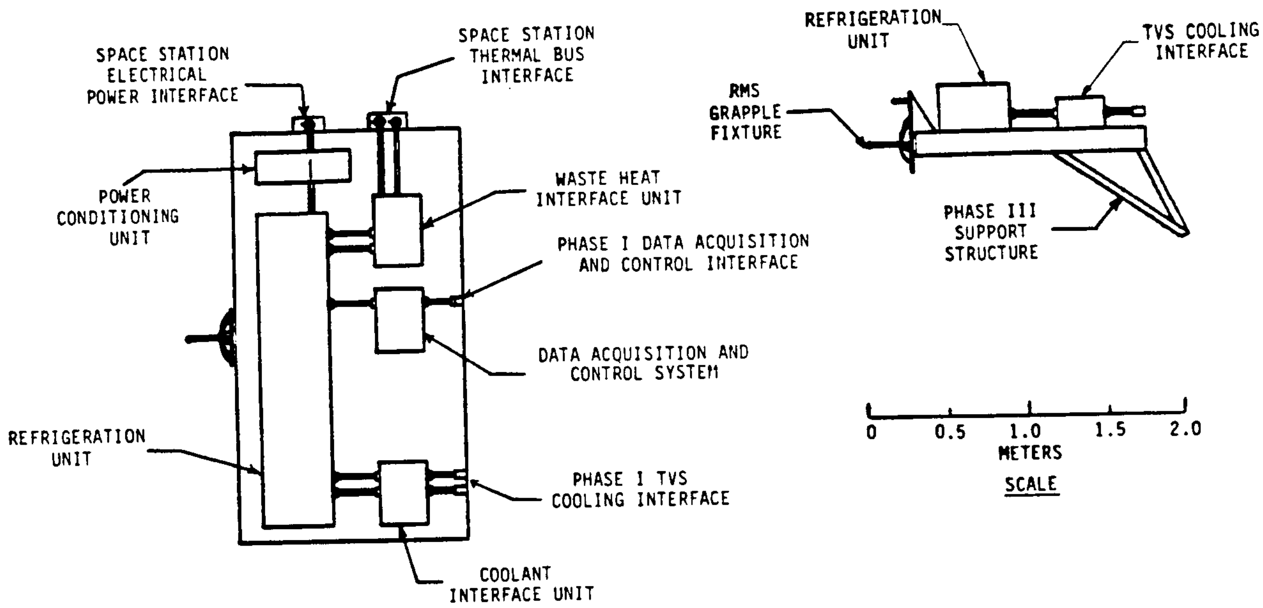


Figure 2-57. PHASE III MODULE - TOP AND SIDE VIEW.

Cooling of the Phase I dewar will be accomplished by passing coolant through fluid lines that run parallel to the Phase I TVS system in the supply tank. This allows the refrigerator to provide all or just a portion of the cooling load required by the supply tank. Thus, a refrigerator system that is sized specifically for the LTCFSE experiment is not required, allowing use of a more economical, "off the shelf", refrigerator. The coolant interface unit will circulate fluid from the cold side of the refrigerator through the TVS system. Gaseous helium will be utilized as the heat transfer fluid due to its inherent safety, superior heat transfer characteristics and its low condensation temperature. Refrigerators that do not circulate cold working fluid, such as the Stirling and Magnetic refrigerators, would use a Coolant Interface Unit (CIU) that consists of a cryogen circulator that would circulate fluid around the cold side of the refrigerator. In other systems that do circulate refrigeration fluid, such as an absorption refrigerator, the CIU would contain a circulator in conjunction with a heat exchanger to interface between the refrigerator coolant and the GHe TVS coolant. It should be noted that these lines would all be MLI wrapped, but are shown exposed in the figures for clarity.

The waste heat interface unit transfers the refrigerator waste heat from the refrigerator hot side to the Space Station thermal cooling bus. The design requirements for different refrigerators are similar to that outlined for the CIU, except that temperatures and heat transfer rates are necessarily higher.

Since the Phase III power requirements (approximately 2.5 kW) are much higher than the previous phases of the experiment, a separate PCU is utilized for the Phase III module. The PCU will provide the required power for both the refrigerator and all Phase III subsystems.

The Phase III DACS interfaces with the Phase I DACS, becoming a subsystem to it. This eliminates the need for a separate Phase III Space Station DACS interface. Controls required for the refrigerator are assumed to be integral within the refrigerator unit. The Phase III DACS contains the waste heat and cooling interface controls and any required sensor conditioning hardware. Phase III hardware specifications are presented in Table 2-XXVIII.

Table 2-XXVIII. PHASE III MODULE EQUIPMENT DESCRIPTION.

Capacity	10 watts at 20 K (36°R)
Heat Rejection Temperature	300 K (540°R)
Heat Rejection Load	2.5 kW
Input Power	2.5 kW
Mass	544 kg (1200 lbm)
Design Lifetime	5 years
Waste Heat HEX area (if req'd)	0.39 m^2 (4.2 ft^2)
Waste Heat HEX effectiveness	0.80
Coolant HEX area (if req'd)	0.20 m^2 (2.2 ft^2)
Coolant circulator capacity	0.025 kg/hr (0.055 lbm/hr)

A fluid flow schematic of the Phase III module interfaced with the Phase I TVS is presented in Figure 2-58. A Stirling cycle refrigerator with hot and cold "fingers" is depicted in the schematic. A Stirling cycle unit was depicted since its high level of development makes it a likely candidate for use in Phase III.

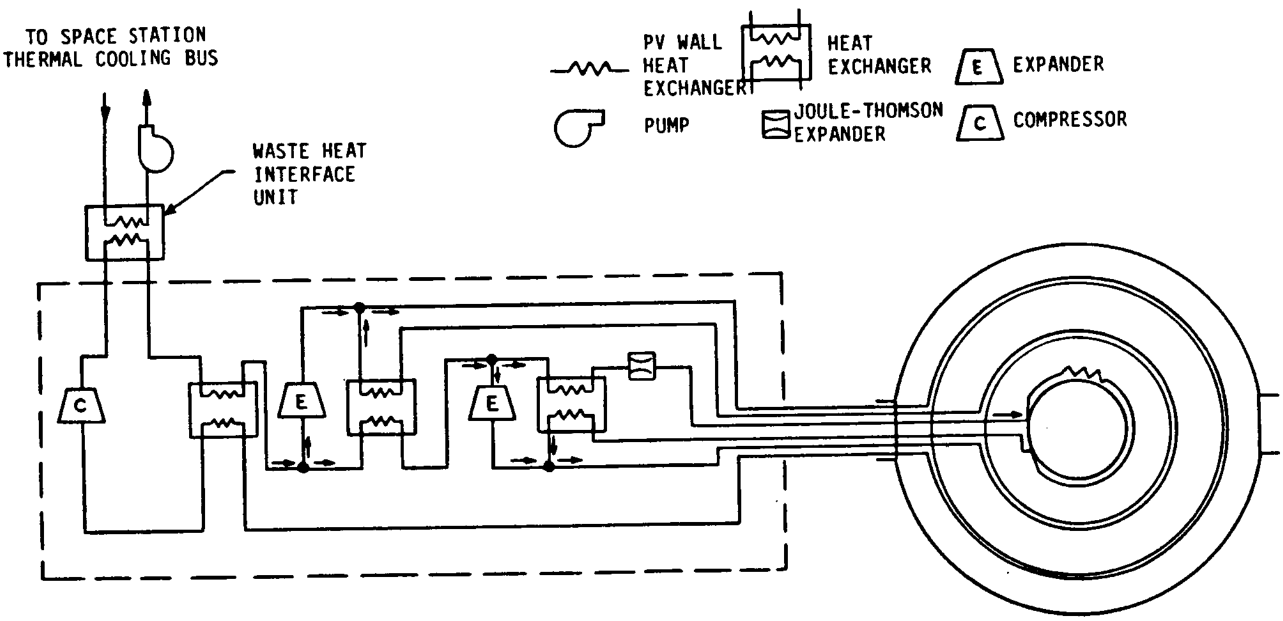


Figure 2-58. PHASE III - SYSTEM SCHEMATIC.

The Phase III instrumentation list is presented in Table 2-XXIX. This instrumentation is in addition to the Phase I and II requirements. Data are to be sampled at a rate of ten times per hour and downlinked weekly. Phase III resource and interface requirements are summarized in Table 2-XXX.

C-2

Table 2-XXIX. PHASE III INSTRUMENTATION LIST.

Coolant Interface Unit Temperatures (4)
Refrigerator Stage Temperatures (3)
Waste Heat Interface Unit Temperatures (4)
Refrigerator Input Power
TVS Coolant Mass Flowrate

Table 2-XXX. PHASE III RESOURCE AND INTERFACE REQUIREMENTS.

RESOURCE REQUIREMENTS:	
Electrical Power - 2.5 kW	
Thermal Bus Cooling Load - 2.5 kW	
Crew Manpower Requirements:	
Deployment/Setup EVA	8 manhours
Deployment/Setup IVA	12 manhours
Data Downlink/Status Check IVA	1 manhour/wk
ADDITIONAL DATA ACQUISITION INTERFACES:	
11 Temperature Transducers	Range 11 to 333 K (20 to 600°R)
Input Power Meter	Range 0 to 2.5 kW
Mass Flowmeter	Range 0 to 0.045 kg/hr (0 to 0.1 lbm/hr)

The Phase III module will be launched on a Shuttle payload bay pallet, preferrably one shared with other hardware, in order to minimize launch costs. Upon reaching Space Station, the module will be deployed on the RMS and then structurally attached to the experiment. During an EVA operation, the fluid and data interfaces between the Phase I and III modules will be connected, as well as the Space Station power and thermal cooling bus interfaces. Phase III hardware operation will then be checked out and verified prior to beginning experimentation. The experiment will be allowed to reach a quasi-steady state condition, after which refrigerator performance will be monitored for a one-year period.

2.4.3 <u>Examination of Potential Experiment Locations.</u> The following locations were considered for the experiment:

1. Free Flyer
2. Tethered to Space Station
3. Space Station Hard Mount

The first option considered was a free flying platform in the vicinity of the Space Station. The advantages and disadvantages of this concept are presented in Table 2-XXXI. The disadvantages inherent in the free-flyer concept precludes it from being a viable location for the long-term storage experiment.

Table 2-XXXI. FREE FLYING PLATFORM - ADVANTAGES AND DISADVANTAGES.

ADVANTAGES	DISADVANTAGES
o Controllable G-level o Minimizes Space Station Safety Issues o Minimizes Space Station Contamination Issues o Minimizes Space Station Resource and Interface Requirements	o Requires on-board power supply and heat rejection system, increaseing experiment weight, complexity and cost o Requires attitude control system o Requires RF data/control link o Reconfiguration more difficult than hard mount o Provides little additional experimental benefit o Highest weight and cost option

The second option considered was a platform connected to Space Station via a tether. The advantages and disadvantages of this concept are presented in Table 2-XXXII. The few advantages and numerous disadvantages inherent in a tethered experiment preclude it from being considered a viable location.

Table 2-XXXII. TETHERED PLATFORM - ADVANTAGES AND DISADVANTAGES.

ADVANTAGES	DISADVANTAGES
o Space Station Power Bus may be utilized o Data/Control Interfaces are simpler than free-flyer o Tethered Concept is lighter and lower cost than free-flyer	o Space Station maneuvering presents problems o Reconfiguration more difficult than hard mount o Utilization of Space Station thermal cooling bus difficult, if not impossible o May require attitude control system o Higher weight and cost than hard mount o Contamination and safety issues are greater than free-flyer

The final option considered was hard-mounting the experiment to the Space Station truss structure. The advantages and disadvantages of this option are presented in Table 2-XXXIII. The hard mount concept has few disadvantages. The increased safety and contamination issues can be readily solved through careful experiment design without greatly increasing experiment cost and complexity or decreasing experiment effectiveness. Based on the numerous advantages and few disadvantages of this concept, a hard-mount has been baselined as the experiment location.

The specific location on Space Station that is recommended is on the lower boom of the current dual-keel station, adjacent to the OTV refueling bay location of the Growth Station. A primary application of the experiment results will be in OTV refueling technology. Placing the experiment adjacent to the OTV bay location will provide an environment identical to that experienced by the OTV tanks. This enhances the applicability of the experiment's results towards OTV refueling technology. In addition, the experiment will have similar interface, contamination, safety and operational issues as the OTV refueling system. Resolution of these issues during experiment development and operation will further enhance OTV refueling system development. Finally, utilization of this location will ensure accessibility of Space Station data, power and thermal bus systems.

Table 2-XXXIII. HARD MOUNT - ADVANTAGES AND DISADVANTAGES.

ADVANTAGES	DISADVANTAGES
o Space Station Power Bus may be utilized	o Increased contamination issues relative to other concepts
o Space Station thermal bus may be utilized	o Increased safety issues relative to other concepts
o Data/Control Interfaces are simplified	o Increases Space Station interface and resource requirements
o Experiment is accessible by the Space Station Remote Manipulator System	
o Assembly problems during reconfiguration are minimized	
o Lowest weight and cost option	

2.5 Task V - Preliminary Evolutionary Plan. The objective of Task V was to develop a preliminary evolutionary plan for the long-term storage experiment that will identify, schedule, and cost all major experiment activities. The primary activities within Task V were as follows:

- o Work Breakdown Structure (WBS) Development
- o ROM Program Costing
- o Program Schedule Development

The following sections summarize these tasks and present the following Task V outputs:

- o Program WBS
- o Program ROM Costs
- o Program Time-Phase Funding
- o Program Schedule

2.5.1 Program Work Breakdown Structure. A preliminary WBS was prepared to provide the elements of cost and schedule for the program plan. Figure 2-59 shows this WBS with detail down to the third level.

The typical subtasks associated with each third level task of Phase I is shown in Figure 2-60, detailing subtasks down to the fifth level. A similar task breakdown is associated with the Phase II and III efforts. Previous Beech program work breakdown structures, including the PRSA WBS, were utilized in preparation of the Long-Term Storage WBS. The WBS is broken down into six major tasks, described below.

Task 1.0 is Program Management, and will last through the duration of the program on a manloaded effort. The Program Manager will be the direct link between NASA-LeRC and the contractor organization. He will implement program plans, direct operations and control schedule and expenditures.

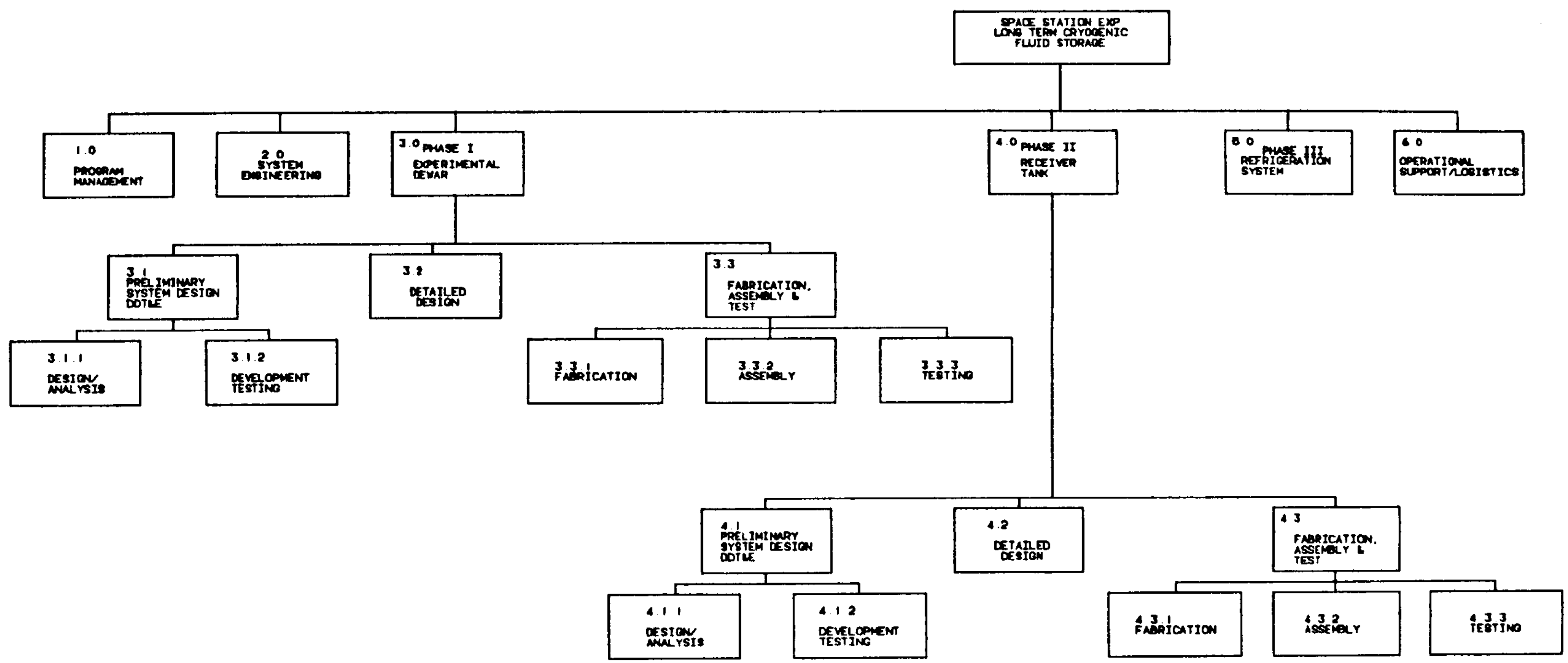


Figure 2-59. WORK BREAKDOWN STRUCTURE.

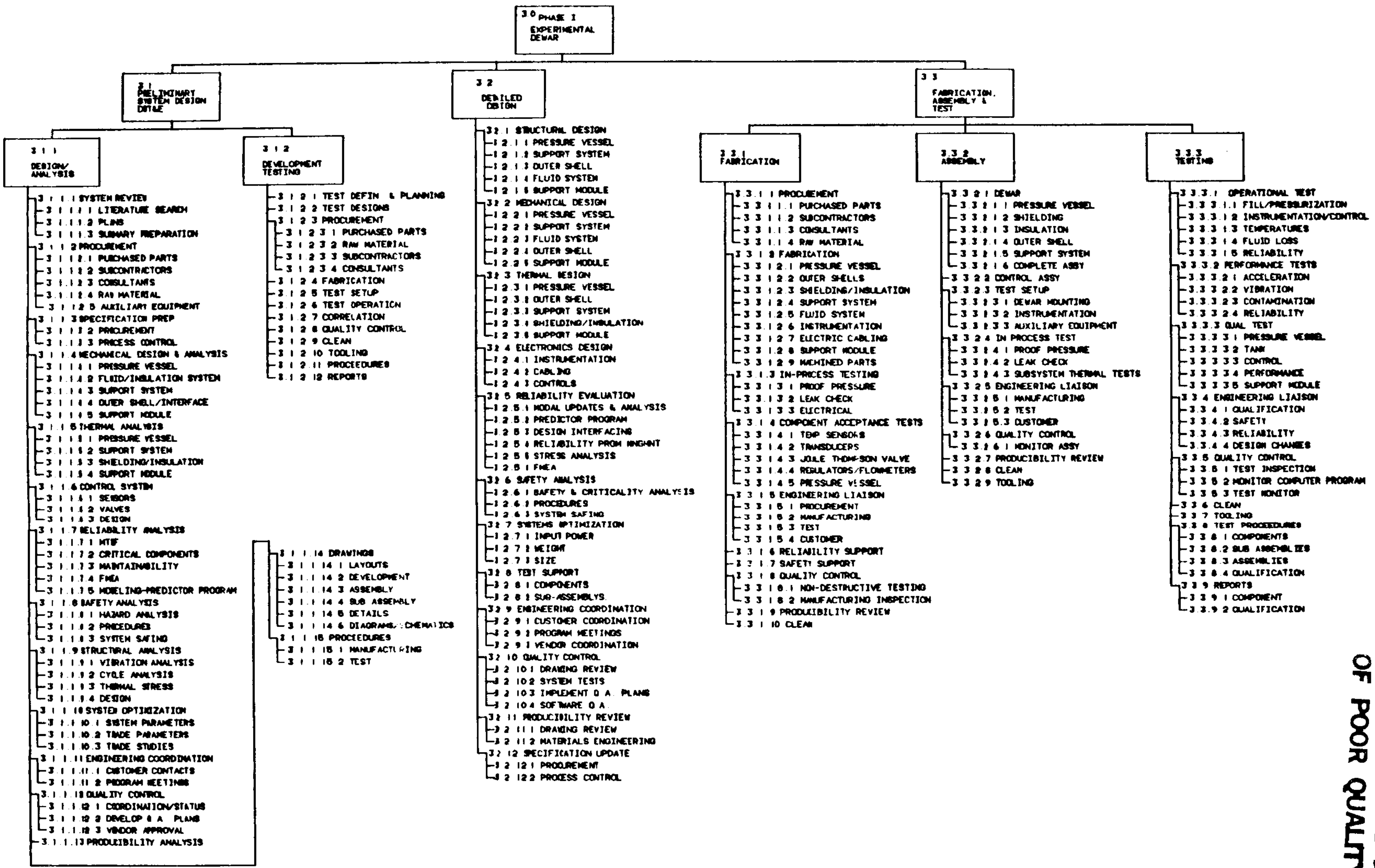


Figure 2-60. PHASE I WBS DETAIL.

Task 2.0 is the Systems Engineering effort. All efforts related to the integration of Phases I, II and III would be performed under this task. Preliminary experiment design, including system performance specifications, will be performed within this task.

Task 3.0 is the production of the Phase I test hardware, including development, design, fabrication, test and assembly. Tasks 4.0 and 5.0 are the production of the Phase II and III hardware, respectively.

Task 6.0 covers the tasks required to support operational and logistics operations. Operational support includes all activities required to support NASA-LeRC during deployment, installation and experiment operation. This also includes all required data reduction and analytical model correlation tasks. Logistics operations include all support required to define spares and to support maintenance and repair of experiment hardware.

2.5.2 Program ROM Costing. Program ROM costing was performed utilizing an existing life cycle cost program developed by Beech Aircraft under contract to NASA-MSFC. This program was developed for MSFC to calculate Space Station cryogenic propellant supply system life cycle costs. The life cycle cost model utilizes PRSA program costs as a basis and separates system costs into eight primary categories:

1. Program Management
2. Design, Development, Test and Engineering (DDT&E)
3. Tooling
4. Qualification
5. Production
6. Maintenance
7. Shuttle Transportation
8. Operations

Suitable modifications were made to the program to make it applicable to the long-term storage experiment. Where the model was not applicable, such as in Phase III costing, a separate costing was performed and added into the eight categories listed above. Table 2-XXXIV shows the resulting ROM costs generated in 1986 dollars. Since there is only one flight article to be built, there were no production costs generated beyond DDT&E costs. No system maintenance is currently baselined for the experiment, thus no maintenance costs were generated. Operations costs include deployment and retrieval EVA/IVA, experiment operation IVA and Space Station electrical power and cooling costs. User costs for the Space Station data acquisition system are currently not available and are not included in the projected operations cost. Technology development costs were not included. However, the cost of applying a particular technology to the system was included. Space Shuttle launch cost was assumed to be $100 million per launch.

Table 2-XXXIV. LTCFSE PROGRAM ROM COSTS.

	COST, 1986, $1000			
CATEGORY	PHASE I	PHASE II	PHASE III	TOTAL
1. Program Management	$ 910	$ 750	$ 750	$ 2,410
2. DDT&E	8,720	7,290	5,720	21,730
3. Tooling	1,430	1,190	210	2,830
4. Qualification Testing	4,890	4,540	2,100	11,530
5. Shuttle Launch	16,580	6,640	8,290	31,510
6. Space Station Operations	5,700	8,770	9,850	24,320
TOTAL PROGRAM COST	$ 38,230	$ 29,180	$ 26,920	$ 94,330

2.5.3 Program Schedule. The overall program schedule is presented in Figure 2-61. The effort labelled "conceptual design" is the current LTCFSE experiment design effort. The number in parenthesis following all subsequent tasks refer to the WBS task number, as depicted in Figure 2-59. All Space Station activities are shown highlighted. This schedule assumes a 1993 deployment date for the IOC Space Station.

2.5.4 Time-Phased Funding. The program time-phased funding is shown in Table 2-XXXV. The funding level for each year is divided into six categories as in Table 2-XXXIV. Program funding begins with Task 2.0, Prelimimary Design, and continues through experiment retrieval.

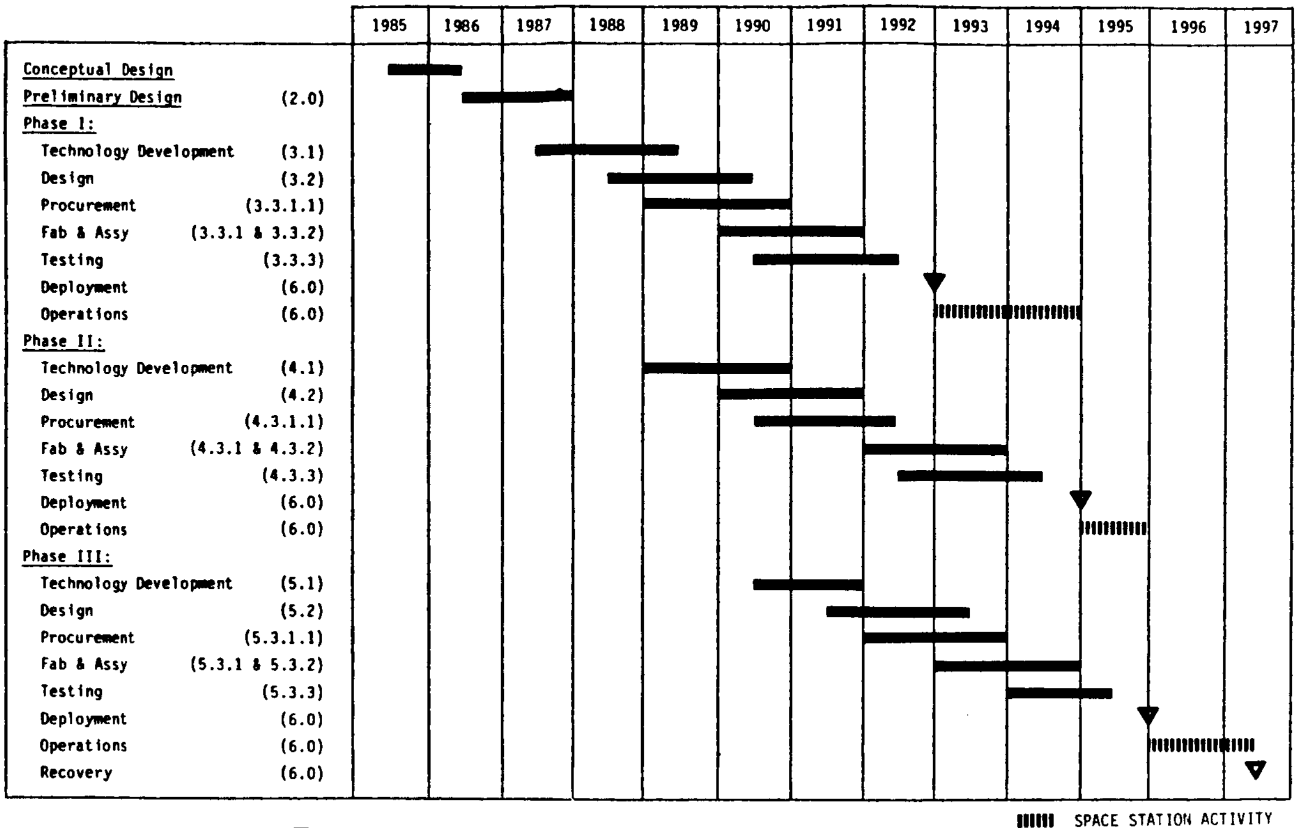


Figure 2-61. LTCFSE PROGRAM SCHEDULE.

Table 2-XXXV. TIME-PHASED FUNDING (1986 $1000).

COST CATEGORY	PGM MGMT	DDT&E	TOOLING	QUAL TEST	LAUNCH	SS OPS	TOTAL
YEAR:							
1986	110	573	---	---	---	---	682
1987	110	1,718	---	---	---	---	1,828
1988	110	1,718	---	---	---	---	1,828
1989	197	2,863	471	---	---	---	3,532
1990	329	4,581	707	1,048	---	---	6,666
1991	329	4,009	471	2,097	---	---	6,906
1992	329	2,291	707	2,097	---	---	5,423
1993	329	2,863	471	2,097	16,582	1,961	24,303
1994	329	1,109	---	3,145	---	1,271	5,855
1995	110	---	---	1,047	6,634	5,752	13,543
1996	110	---	---	---	8,291	7,475	15,876
1997	22	---	---	---	---	7,859	7,881
TOTAL	$2,414	$21,725	$2,827	$11,533	$31,506	$24,318	$94,323

3.0 CONCLUSIONS AND RECOMMENDATIONS

3.1 Conclusions. The Long Term Cryogenic Fluid Storage Experiment defined by this study, provides demonstration and evaluation of the critical technologies required by future orbital cryogenic systems. Such systems include those required for the space-based OTV and those required for Space Station user and life support (liquid nitrogen). It is, therefore, imperative that Low-G Long Term Cryogenic Fluid Storage be understood with respect to thermal performance, fluid acquisition and transfer issues. This is precisely what the LTCFSE is designed to do.

The general approach taken in the design of the experiment has been to divide it into three phases. Not only did the critical technologies to be demonstrated seem to fall naturally into three categories but this phased approach has added benefits. Being able to begin deployment on the experiment sooner and making use of knowledge gained from previous phases are two advantages. Another benefit from this phased approach was the evolution of its modular design. Modularity simplifies space station logistics during configuration changes and promotes the possible multiple use of the experimental facility. The modular design and relatively small size of the LTCFSE allows it to be ground refurbished for additional testing at a minimal cost. Experiment modularity provides maximum flexibility for potential future uses of the experiment, either for further testing or for practical use aboard Space Station. For example, resupply for the experiment could extend experiment lifetime or allow the supply tank to be used for Space Station cryogen supply, or the hardware could be used with a scaled "dummy" OTV vehicle to demonstrate refueling operations.

It is important to emphasize that the LTCFSE is viewed as an experiment rather than a technology demonstration. Its real value lies in the thermal performance data it will be generating. The long-term nature of LTCFSE experimentation will provide a large data base of information, including effects of extended exposure to the orbital environment. The large amount of data that will be gathered will be invaluable for correlation of both general purpose low-g fluid and thermal models. Correlation of these models, (with respect to fluid dynamic and geometric parameters), will allow future designs of OTV and other orbital systems to be performed with more certainty.

Two other points worth emphasizing here are of a more practical nature: the use of existing hardware and the proposed location of the experiment itself. Utilization of existing hardware and designs minimizes experiment development costs. Table 3-I lists the existing hardware that is recommended for use on the LTCFSE along with their recommended use. It should be noted that the OTTA and ELMS tanks require modifications prior to use on the experiment. An examination of required modifications and the condition of the hardware may indicate that use of a portion of the hardware or only the design is more cost effective. In such a case, use of the existing design and tooling will still provide cost savings.

Table 3-I. AVAILABLE HARDWARE RECOMMENDATIONS.

HARDWARE	RECOMMENDED USE
Oxygen Thermal Test Article (OTTA) Tank	Phase I Supply Tank
Earth Limb Measurement Satellite (ELMS) Tank	Phase II Receiver Tank
Fuel Cell Servicing System (FCSS)	Supply Tank LH_2 Loading
Centaur Orbiter Mod Kit	Phase I Flight Vent & Dump System

Development and qualification of the supply and receiver tanks would provide two different sized Shuttle and Space Station qualified designs. The design could be readily modified and requalified to store other fluids. The high cost required to develop such flight-qualified tanks makes them a valuable resource for future use.

Locating the LTCFSE on Space Station adjacent to the proposed OTV servicing bay location will provide further benefits. The LTCFSE will experience the same environment as the future OTV facility. This will allow assessment of environmental effects prior to deployment of these facilities. Deployment and operation of the LTCFSE will provide insight into any problems associated with the deployment and operation of these future facilities. The resource and interface requirements of this experiment are also similar to the OTV facility. Early definitions of these requirements will allow Space Station Phase C/D design to accommodate them, thus ensuring these capabilities are in place for use by the proposed OTV servicing facility.

Several times in the course of this report, the Cryogenic Fluid Management Flight Experiment (CFMFE) has been mentioned. As CFMFE is currently the only funded flight experiment which addresses many of the same critical technologies and issues identified in this report, it would be interesting to draw some comparisons. Table 2-I presented the hardware required and Table 2-II presented the technology issues that must be addressed in the design of Space Station OTV systems. The LTCFSE would provide the only on-orbit demonstration of many of these required technologies. The technologies that are uniquely demonstrated by the LTCFSE are summarized in Table 3-II, and those technologies shared by the two experiments are summarized in Table 3-III.

Table 3-II. TECHNOLOGIES UNIQUELY DEMONSTRATED BY THE LTCFSE.

PASSIVE THERMAL:	FLUID TRANSFER:
Dual Stage Support	Low Heat Leak Valves
Para-Ortho H_2 Conversion	Low Heat Leak Transfer Lines
Thermal Coatings	Cryogenic Disconnects
	Boiloff Collection for External Pressurization
	Metal Hydride Compressor
INVESTIGATED PHENOMENA:	**ACTIVE REFRIGERATION:**
Long-Term Stratification	Long Lifetime Refrigerator
S.O.S Performance Degradation	Cryogenic Heat Exchanger
Thermal Coating Degradation	Cryogenic Circulator
Micrometeroid Protection	Refrigerator to S.S. Thermal Bus HEX

Table 3-III. TECHNOLOGIES DEMONSTRATED BY BOTH THE LTCFSE AND THE CFMFE.

PASSIVE THERMAL:	FLUID TRANSFER:
Thick MLI	Capillary Acquisition
TVS	Low-G Quantity Gaging
Soft Outer Shell	Mass Flow Meters
Hard Outer Shell	HPG Pressurization

In addition to the above mentioned difference, the LTCFSE is (i) uniquely a long term storage experiment and (ii) it will provide added data at different m/V's for the supply and receiver tanks. As a final point, although the LTCFSE demonstrates some of the same technologies as CFMFE, it provides an opportunity to further develop these technologies using the experience gained from the CFMFE program.

3.2 Recommendations. It is recommended that a follow-on design effort be implemented with the following objectives:

1. Prepare a detailed description of experimental objectives based on the LTCFSE conceptual design.

2. Advance the Phase I design one step further, producing a layout drawing of the design.

3. Perform more detailed thermal, fluid and structural analysis of the Phase I design.

4. Prepare a Phase I design specification.

5. Present the design concept to NASA-Johnson Space Center payload integration and safety personnel for design inputs.

6. Update the MRDB and the TDAG forms as required.

Performing the above recommendations will ensure the experimental requirements are as complete and as firm as possible prior to entering a detailed effort. Preparation of detailed objectives will allow further definition of instrumentation requirements and definition of the analytical models for correlation to test data. Comprehensive definition of modelling efforts further enhances instrumentation selection, as does performing the detailed analysis outlined above in objective number three. For example, a primary objective of Phase I is to demonstrate and evaluate thermodynamic vent system performance. Defining this objective in detail and performing thermal and fluid analysis of this system will ensure that the definition of instrumentation is adequate.
Performing a more detailed design of the LTCFSE also ensures that accurate Space Station resource and interface requirements are defined. Accurate inputs to Space Station Phase C/D design personnel will ensure the Space Station design will accommodate both the LTCFSE and OTV servicing facility requirements. Performing a more detailed design of the LTCFSE also ensures that accurate Space Station resource and interface requirements are defined. Accurate inputs to Space Station Phase C/D design personnel will ensure the Space Station design will accommodate both the LTCFSE and OTV servicing facility requirements.

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APPENDIX A

TECHNOLOGY ASSESSMENTS

TITLE: Stirling Cycle Refrigerators

GENERIC CATEGORY: Thermal Performance - Refrigerators

TECHNOLOGY ELEMENTS:

Magnetic bearings, linear induction, drive motors, clearance seals.

FLIGHT EXPERIENCE:

Phillips Magnetic Bearing Stirling Cycle Refrigerators - 4 units flown in 1979 to cool Gamma Ray Spectrometer Detectors.

ADVANTAGES:

Performs well at low heat loads, technology rapidly maturing, relatively high efficiencies.

DISADVANTAGES:

Long-term performance ($\geq$ 5 years) and reliability not yet demonstrated.

SYSTEM LEVEL DEMONSTRATIONS:

See Table A-I.

DEMONSTRATED PERFORMANCE:

See Table A-I.

DEMONSTRATED RELIABILITY:

See Table A-I.

PROBLEM AREAS:

Long-term life and reliability, working fluid contamination and leakage. Vuilleumier-Material wear and fatigue.

POSSIBLE IMPROVEMENTS:

Development of long lifetime units, high heat capacity regenerators, and materials with longer lifetime and higher reliability.

TECHNOLOGY ASSESSMENT:

Technology rapidly developing, many development programs are currently in progress.

RISK ASSESSMENT:

Long-term reliability and performance are currently the biggest risk items. Development of long lifetime Stirling cycle refrigerator entails lowest risk of all refrigerator technologies due to the large amount of technology development that has been performed.

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7. Bradshaw, T. W., et al, "Performance of the Oxford Miniature Stirling Cycle Refrigerator", presented at the 1985 Cryogenic Engineering Conference.
8. Werrett, S., T., et al, "Development of a Small Stirling Cycle Cooler for Spaceflight Applications", presented at the 1985 Cryogenic Engineering Conference.
9. Sherman, A., et al, "Progress on the Development of a 3-5 Year Lifetime Stirling Cycle Refrigerator for Space", Advances in Cryogenic Engineering, Vol. 25.

10. Maiden, T. E., "Cryogenic Closed Cycle Refrigerators", Beech Aircraft MR-17901.

11. Sherman, A., History, "Status and Future Applications of Spaceborne Cryogenic Systems", Advances in Cryogenic Engineering, Vol. 27.

12. Harris, R. E., et al, "Cryo-Cooler Development for Space Flight Applications".

13. Daniels, A., et al, "Magnetically Suspended Stirling Cryogenic Space Refrigerator: Test Results Advances in Cryogenic Engineering", Vol. 29.

14. Fester, D. A., et al, "Proceedings of the Long-Term Cryogenic Storage Technology Conference", May 12-13, 1982, Martin Marietta Aerospace.

Table A-I. STIRLING CYCLE REFRIGERATORS.

	PHILLIPS RHOMBIC DRIVE STIRLING	PHILLIPS/GSFC LINEAR DRIVE STIRLING MAGNETIC BEARING MOVING MAGNET	LINEAR DRIVE MOVING COIL FREE PISTON	MAGNAVOX 2-STAGE STIRLING LINEAR MOTOR MAGNETIC BEARING
CAPACITY/TEMP (w)/(K)				
Stage 1 Stage 2 Stage 3	1.5/143 0.3/67	5/65	5/65	10/Classified 5/Classified
Input Power, W	35	220	155	700
Weight, kg (Refrigerators & Electronics)	7.2	62	2	82
Demonstrated Lifetime, hrs	8100	6500	5000	---
Flight Time, hrs	15,000 DOD P-78-1 Spacecraft 1979 Launch (4 units)	---	---	---
Comments	Performance Degraded to 50% of Initial Performance During Lifetime		Slight Degradation in Performance after 5000 hrs	

Table A-I. STIRLING CYCLE REFRIGERATORS (concluded).

	OXFORD SPLIT STIRLING CYCLE, CLEARANCE SEALS/SPIRAL SPRINGS	HUGHES-VUILLEUMIER CYCLE (STIRLING CYCLE CYCLE W/THERMAL COMPRESSOR (HI-CAP)
CAPACITY/TEMP (w)/(K)		
Stage 1 Stage 2 Stage 3	.875/80	12/68 8.3/64 10/39 1.9/14 .3/14 .15/10
Input Power, W	30	2100
Weight, kg (Refrigerators & Electronics)	5 (no electronics)	82
Demonstrated Lifetime, hrs	6000 (no change in performance)	49,000 total (for 3 units)
Flight Time, hrs	---	---
Comments	Will fly on NASA Upper Atmospheric Research Satellite in late 1989 Design Life - 3yrs	Will be flown on Space Infrared Sensor (SIRE)

TITLE: Absorption Refrigerators

GENERIC CATEGORY: Thermal Performance - Refrigerators

TECHNOLOGY ELEMENTS:

Absorption/adsorption compressors, Joule-Thomson valves, check valves, thermal switches, heat exchangers.

FLIGHT EXPERIENCE:

None.

ADVANTAGES:

Few moving parts (check valves) provides potential for high reliability/long life. Power input can be electrical, or a direct heat source, such as waste heat, solar, or a radioisotope thermoelectric generator (RTG), such as the SP-100 being developed.

DISADVANTAGES:

Space applications would require low temperature radiators in intermediate stages, low efficiency.

SYSTEM LEVEL DEMONSTRATIONS:

None.

DEMONSTRATED PERFORMANCE:

A complete system at JPL using a hydrogen working fluid and $LaNi_5$ compressors, produced one watt of cooling from 14-29K, with an input power of 400 W.

DEMONSTRATED RELIABILITY:

The above system has been operated for over 1000 hours. The $LaNi_5$ compressor has been operated separately for over 5800 hours. The check valves have been pressure cycled 86 million times (equivalent to 500 years of life in an absorption system).

PROBLEM AREAS:

Joule-Thomson valve contamination.

KEY ISSUES:

Life of metal hydride in compressor, development of thermal switches for use with constant heat source (solar, RTG), power requirements, system weight.

POSSIBLE IMPROVEMENTS:

Improvements in efficiency, operation using constant heat source.

TECHNOLOGY ASSESSMENT:

Technology new and undeveloped, but has high potential to produce a very long lifetime refrigerator.

RISK ASSESSMENT:

Technology is currently high risk until key issues are resolved and an adsorption refrigeration suitable for space flight has undergone long-term testing.

REFERENCES:

1. Jones, J. A. and Golben, P. M., "Life Test Results of Hydride Compressors for Cryogenic Refrigerators", AIAA Paper No. 84-0058. Presented at the AIAA 22nd Aerospace Sciences Meeting, January 1984.
2. Chan, C. K., et al, "Miniature J-T Refrigerators using Adsorption Compressors", Advances in Cryogenic Engineering, Vol. 27, Plenum Press, New York (1982), pp. 735-743.
3. Garrison, P. W., "Molecular Absorption Cryogenic Cooler for Hydrogen Tank Thermal Control", Proceedings of the Long-Term Cryogenic Storage Conference, MCR-82-561, Martin Marietta Aerospace, May 12-13, 1982, pp. 237-256.
4. Fester, D. A., et al", Long-Term Cryogenic Storage Study - Interim Report", AFRPL TR-82-077, prepared by Martin Marietta Aerospace for the Air Force Rocket Propulsion Laboratory.

TITLE: Brayton Cycle Refrigerators

GENERIC CATEGORY: Thermal Performance - Refrigerators

TECHNOLOGY ELEMENTS:

Gas bearings, turbine compressors and expanders, heat exchangers.

FLIGHT EXPERIENCE:

None.

ADVANTAGES:

No metal-to-metal contact due to gas film bearings. Good potential for long life. No reciprocating components, little vibration, detached cold section, wide load range, proven component technology.

DISADVANTAGES:

Turbo-Brayton cycles are inefficient at low gas flows (low heat loads).

SYSTEM LEVEL DEMONSTRATIONS:

None.

DEMONSTRATED PERFORMANCE:

Airesearch has demonstrated a two-stage refrigerator that provided 5 & 20 W of cooling power at classified temperatures. The refrigerator required 2300W of input power and weighed 91kg, including electronics.

DEMONSTRATED RELIABILITY:

System has been operated for approximately 1000 hours. System reliability has been calculated to be between 0.94 and 0.97 for a continuous operating life of three years. Compressor was subjected to 500 start/stop cycles with no discernable wear.

PROBLEM AREAS:

Leakage and contamination of working fluid, reliability of control electronics. Reliability of gas bearings, compressor motor life.

KEY ISSUES:

System life demonstration, electronics reliability, degradation due to leakage and contamination.

POSSIBLE IMPROVEMENTS:

Improve reliability of control electronics. Improve design and manufacturing techniques to minimize leakage and contamination.

TECHNOLOGY ASSESSMENT:

Technology developing gas film bearing compressor and expander technology is well proven, system level technology is major development issue.

RISK ASSESSMENT:

Development entails moderate risk, mostly at the system level.

REFERENCES:

1. Johnson, A. L., "Spacecraft Borne Long Life Cryogenic Refrigerator Status and Trends", Cryogenics, July 1983.
2. Harris, Roberg E., et al, "Cryo-Cooler Development for Space Flight Applications".
3. Maiden, T. E., "Cryogenic Cooler-Cycle Refrigerators", Beech Aircraft MR-17801.
4. Fester, D. A., et al, "Long-Term Cryogenic Storage Study", AFRPL TR-82-071, performed by Martin Marietta Aerospace for the Air Force Rocket Propulsion Lab.
5. Buchmann, O., "Airesearch Cryogenic Turbo-Refrigerator Characteristics", Proceedings of the Long-Term Cryogenic Storage Technology Conference held May 12-13, 1982, at Martin Marietta Aerospace, Denver Division.

TITLE: Magnetic Refrigerators

GENERIC CATEGORY: Thermal Performance - Refrigerators

TECHNOLOGY ELEMENTS:

Solid magnetic working material, superconducting magnet, superinsulated dewar, working material drive motor, heat exchange fluid, fluid drive pump, heat exchanger.

FLIGHT EXPERIENCE:

None.

ADVANTAGES:

High efficiency (over 50% of Carnot efficiency possible - 6x increase over gas refrigeration), high reliability (few moving parts, slow movement), low weight and volume. Can operate with high efficiency at extremely low temperatures (<10K).

DISADVANTAGES:

Superconducting magnets needing cryogenic cooling are required, technology in low state of development.

SYSTEM LEVEL DEMONSTRATIONS:

None.

DEMONSTRATED PERFORMANCE:

A magnetic refrigerator operating near room temperature has been built by Los Alamos Laboratory. Hughes Aircraft, in conjunction with Los Alamos National Laboratories, is currently developing a multistage refrigerator for operation in the 4-60K temperature range. Predicted performance for magnetic refrigerators is shown in Table A-II. A comparison of magnetic refrigerator vs. gas cycle refrigerator efficiency is presented in Figure A-1.

DEMONSTRATED RELIABILITY:

None.

PROBLEM AREAS:

A single stage of a magnetic refrigerator operates over a smaller temperature range then gas cycle refrigerators.

KEY ISSUES:

Development and demonstration of a cryogenic temperature refrigerator.

POSSIBLE IMPROVEMENTS:

Cascading of refrigerator stages to allow operation over a wide temperature range.

TECHNOLOGY ASSESSMENT:

Technology is still in early research and development phase. Long-term development program required to mature technology.

RISK ASSESSMENT:

As this technolgy is in its infancy, development entails a high risk, yet one that would reap large benefits due to the inherent high efficiency of such a system.

REFERENCES:

1. Fester, D. A., et al, "Long-Term Cryogenic Storage Study", AFPRL TR-82-071, performed by Martin Marietta Aerospace for the Air Force Rocket Propulsion Laboratory.
2. Maiden, T. E., "Cryogenic Closed Cycle Refrigerators", Beech Aircraft MR-17801.
3. Barclay, J. A., Los Alamos Scientific Laboratory, letter to R. Scarlotti.
4. Steyert, W. A., "Small Magnetic Refrigerators to Pump Heat from Helium Temperatures to Above 10K", Applications of Closed-Cycle Cryocoolers to Small Superconducting Devices, NBS-SP508.

5. Barclay, J. A., et al, "Magnetic Refrigeration Systems Applicable to Space-Based Sensors", AFWAL-TR-85-3029, Performed by Los Alamos National Laboratory for the Air Force Wright Aeronautical Laboratories.
6. Barclay, J. A., et al, "Magnetic Refrigeration for 4-20K Applications", AFWAL-TR-83-3120, Performed by Los Alamos National Laboratory for the Air Force Wright Aeronautical Laboratories.
7. Mastrup, F. N., "Multistage Magnetic Refrigerator Development; Phase I", AFWAL-TR-83-3079, Performed by Hughes Aircraft Company for the Air Force Wright Aeronautical Laboratories.

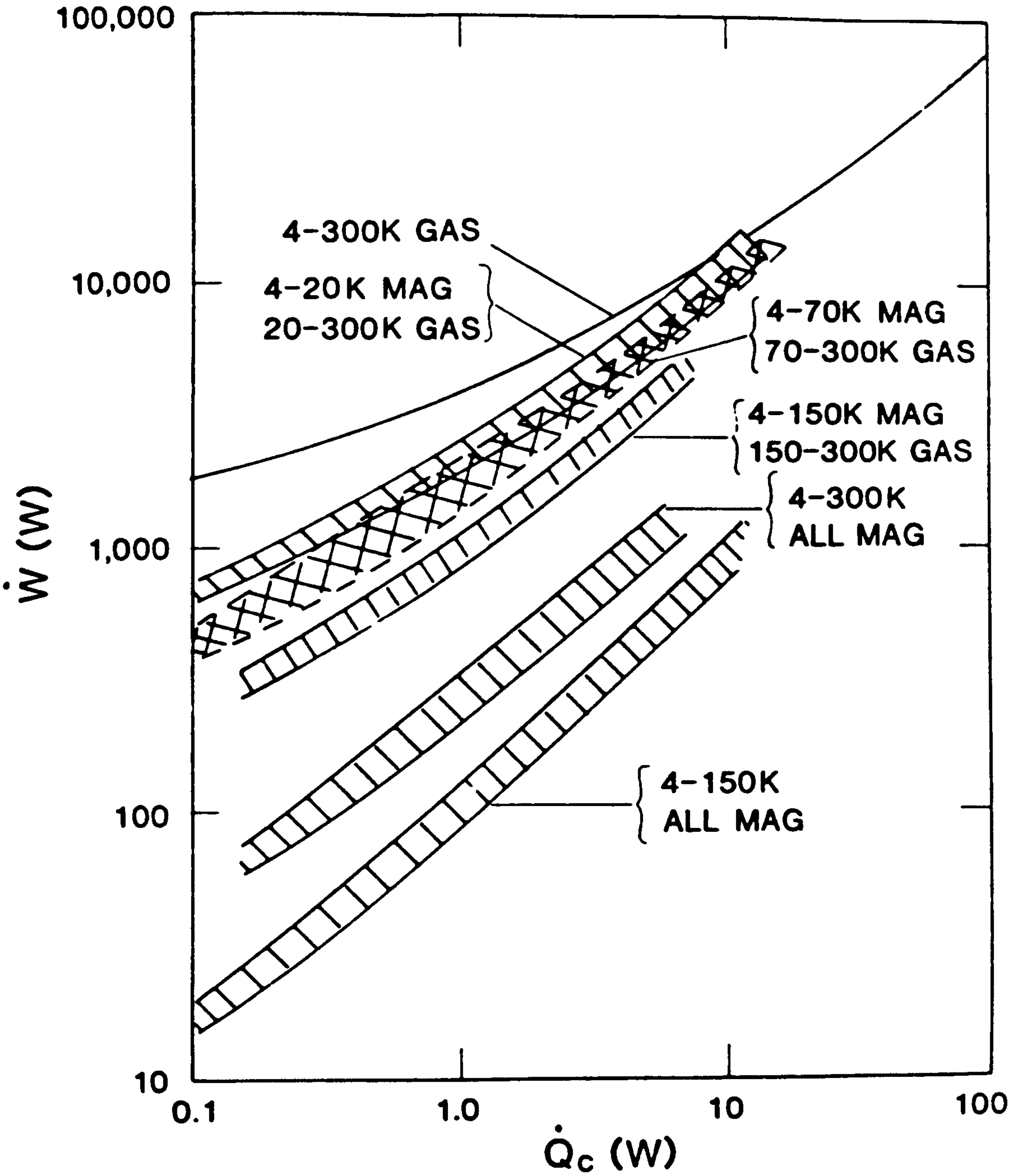


Figure A-1 (Reference 5).
INPUT POWER AS A FUNCTION OF COOLING POWER FOR DIFFERENT COMBINATIONS OF GAS AND MAGNETIC REFRIGERATORS THAT SPAN 4K TO 300K.

Table A-II (Reference 3).
PREDICTED MAGNETIC REFRIGERATORS PERFORMANCE.

SMALL SYSTEM:

Cooling Power: 3W (20.4 Btu/hr) @ 22K, 6W (10.2 Btu/hr) @ 150K
Volume of Magnetic Material: lower stage 68 cm^3, upper stage 195 cm^3
Total Volume of Refrigerator: 37.1 L ($3.71 \times 10^{-2} m^3$, 1.31 ft^3)
Total Mass of Refrigerator: 82.4 kg (182 lbs)
Total Input Power Required: 198 w (0.27 HP)
Power Rejection Required @ 300K: 204 w
Overall Efficiency: 41% of Carnot (22K - 300K)

MEDIUM SYSTEM:

Cooling Power: 4 W (29.0 Btu/hr) @ 22K, 8.5W (12.8 Btu/hr) @ 150K
Volume of Magnetic Material: lower stage 117 cm^3, upper stage 274 cm^3
Total Volume of Refrigerator: 48.0 L ($4.80 \times 10^{-2} m^3$, 1.70 ft^3)
Total Mass of Refrigerator: 104 kg (230 lbs)
Total Input Power Required: 275 w (0.37 HP)
Power Rejection Required @ 300K: 284 w
Overall Efficiency: 42% of Carnot (22K - 300K)

LARGE SYSTEM:

Cooling Power: 8W (81.9 Btu/hr) @ 22K, 24W (27.3 Btu/hr) @ 150K
Volume of Magnetic Material: lower stage 260 cm^3, upper stage 495 cm^3
Total Volume of Refrigerator: 76.3 L ($7.63 \times 10^{-2} m^3$, 2.70 ft^3)
Total Mass of Refrigerator: 166 kg (367 lbs)
Total Input Power Required: 738 w (1.0 HP)
Power Rejection Required @ 300K: 772 w
Overall Efficiency: 44% of Carnot (22K - 300K)

TITLE: Rotary Reciprocating Refrigerators

GENERIC CATEGORY: Thermal Performance - Refrigerators

TECHNOLOGY ELEMENTS:

Gas film bearings, electromagnetic drive, high effectiveness heat exchangers.

FLIGHT EXPERIENCE:

None.

ADVANTAGES:

High efficiency, long development history (since 1962), detached cold section allows for easy integration with heat loads, no metal-to-metal contact within moving parts, good life potential.

DISADVANTAGES:

High weight, complex machining, complex control circuitry, requires high effectiveness heat exchangers.

SYSTEM LEVEL DEMONSTRATIONS:

None.

DEMONSTRATED PERFORMANCE:

A. D. Little Two-Stage System: 1.2W @ 12K, 40W @ 60K; input power - 2670W; weight - 210kg.

DEMONSTRATED RELIABILITY:

A. D. Little: compressor - 9085 hrs; expander - 6557 hrs; system - 6498 hrs.

PROBLEM AREAS:

Contamination of gas film bearings, working fluid retention.

KEY ISSUES:

Development of gas film bearings, fabrication of heat exchangers, contamination control.

POSSIBLE IMPROVEMENTS:

Improved contamination control.

TECHNOLOGY ASSESSMENT:

Technology relatively mature.

RISK ASSESSMENT:

Due to long history of development, this technology has a relatively low development risk.

REFERENCES:

1. Maiden, T. E., "Cryogenic Closed Cycle Refrigerators", Beech Aircraft MR-17801.
2. Harris, R. E., et al, "Cryo-Cooler Development for Space Flight Applications".
3. Johnson, A. L., "Spacecraft Borne Long Life Cryogenic Refrigeration Status and Trends".
4. Smith, Joseph L., et al, "Survey of the State-of-the-Art of Miniature Cryo Coolers for Superconductive Devices", Prepared by the MIT Cryogenic Engineering Laboratory for the Office of Naval Research, January 1984.
5. White, R. and Haskin, W., "Development Approaches for Long-Life Cryo-Coolers", Refrigeration for Cryogenic Sensors and Electronic Systems, NBS SP607.

TITLE: Para to Ortho H_2 Conversion

GENERIC CATEGORY: Thermal Performance

TECHNOLOGY ELEMENTS:

Catalyst bed.

FLIGHT EXPERIENCE:

None.

ADVANTAGES:

Effective use of the endothermic para to ortho conversion increases the cooling capability of hydrogen by approximately 10% as it boils or sublimes and rises to room temperature.

DISADVANTAGES:

Applications of technology not yet developed.

SYSTEM LEVEL DEMONSTRATIONS:

No system level demonstrations of component cooling ability. However, ortho-para converters are used in all H_2 liquefaction plants on a system level.

DEMONSTRATED PERFORMANCE:

Numerous demonstrations of para to ortho conversion have been performed to study effects of flowrate, temperature, pressure and type of catalyst bed. To date, none have provided a demonstration of practical applications, such as cooling a dewar through use of a vapor cooled shield, heat station, or component cooling. Lockheed has performed testing on the effectiveness of a catalyst bed utilizing Apachi-I catalyst. This test measured effectiveness versus flowrate and temperature. Both liquid and solid hydrogen were used as a source of para hydrogen (Reference 2).

DEMONSTRATED RELIABILITY:

The use of a catalyst bed for Ortho-Para conversion has performed reliably for long-term in hydrogen liquefaction plants. As the same catalyst can be used in Para-Ortho conversion, its use can be said to be proven reliable over long-term use.

PROBLEM AREAS:

Catalyst contamination.

KEY ISSUES:

Prevention of catalyst contamination, integration on a system level to produce useful cooling.

POSSIBLE IMPROVEMENTS:

Development of system level cooling demonstration.

TECHNOLOGY ASSESSMENT:

Catalyst bed/conversion technology is mature. Technology needs to be developed and matured in terms of practical cooling applications.

RISK ASSESSMENT:

Development towards practical applications would incur minimal risk.

REFERENCES:

1. Sherman, A., "Cooling by Para-to-Ortho Hydrogen Conversion", GSC-12770, NASA Tech Briefs, Vol. 7, No. 3, Spring 1983.
2. Nast, T. C. and Hsu, I. C., "Development of a Para-Ortho Hydrogen Catalytic Converter for a Solid Hydrogen Cooler", Advances in Cryogenic Engineering, Vol 29, Plenum Press, 1984, pp. 723-731.
3. Clark, R. G., et al, "Investigation of the Para-Ortho Shift of Hydrogen", ASD TDR-62-833, prepared by Air Products and Chemicals, Inc., for the Air Force Aero-Propulsion Laboratory.
4. Singleton, A. H., "A Rate Model for the Low Temperature Catalytic Ortho-Para Hydrogen Reaction", Doctoral Thesis, Lehigh University, 1968.
5. Singelton, A. H. and Lapin, A., "Design of Para-Ortho Hydrogen Catalytic Reactors", Advances in Cryogenic Engineering, Vol. II, Plenum Press, 1966, pp. 617-630.

TITLE: Passive Radiators - Cryogenic

GENERIC CATEGORY: Thermal Performance - Radiators

TECHNOLOGY ELEMENTS:

Radiator surface, heat pipes.

FLIGHT EXPERIENCE:

Cryogenic radiators have flown on several Department of Defense (DOD) spacecraft.

ADVANTAGES:

Reliability, simplicity, no power consumption, no vibration, mature technology.

DISADVANTAGES:

Low cooling capacity at cryogenic temperatures, cooling below 70K impractical, constraints imposed on vehicle orientation.

SYSTEM LEVEL DEMONSTRATIONS:

Several radiators have performed on a system level on DOD satellites, details classified.

DEMONSTRATED PERFORMANCE:

See Table A-III for a summary of ground test radiator performance. Several one and two stage radiators with cooling capacities from 1 to 10 mW at 90 to 120K have performed on-orbit.

DEMONSTRATED RELIABILITY:

Radiators have functioned reliably for long-term on several DOD satellites.

PROBLEM AREAS:

Dependency on vehicle orientation, reduction of parasitic heat leak. Optical properties degradation and micro meteroid damage of radiator surface.

KEY ISSUES:

Development of thermal diode heat pipes and low heat leak insulation/support systems.

POSSIBLE IMPROVEMENTS:

Lowering of parasitic heat leak would improve radiator performance. Integration with thermal diode heat pipes would reduce constraints on spacecraft attitude.

TECHNOLOGY ASSESSMENT:

Current SOA radiators operate within 10-20% of theoretical capacity. Technology is well developed.

RISK ASSESSMENT:

Low risk involved in use of cryogenic radiators.

REFERENCES:

1. Fester, D. A., et al, "Long-Term Cryogenic Storage Study", AFPRL-TR-82-071, performed by Martin Marietta Aerospace for the Air Force Rocket Propulsion Laboratory.
2. Haskin, W. L. and Dexter, P. F., "Ranges of Application for Cryogenic Radiator and Refrigerators on Space Satellites", Proceedings AIAA 17th Aerospace Sciences Meeting.

Table A-III (Reference 1).
DEMONSTRATED PERFORMANCE: PASSIVE RADIATORS.

RADIATOR	AREA m^2 (Ft^2)	OPERATING TEMP, K (°R)	HEAT LOAD, W (BTU/hr)	HEAT PIPES	ORBIT
RM-20A	1.1 (11.8)	100 (180)	2 (6.8)	CH_4	SUN-SYNCH
RM-20B	1.3 (14.0)	138 (248)	12 (41.0)	FREON, CH_4	SUN-SYNCH
CRTU 2-STAGE	8.0 (86.1) 6.8 (73.2)	100 (180) 70 (126)	0 5 (17.1)	O_2	GEO-SYNCH
CRTU 3-STAGE	8.0 (86.1) 6.8 (73.2) 5.1 (54.9)	118 (212) 63 (113) 41 (74)	0 0 0.24 (0.82)	O_2	GEO-SYNCH
ANGLED-SHIELD	4.0 (43.1)	70 (126)	5 17.1)	-- (ESTIMATE)	GEO-SYNCH
ANGLED-SHIELD	0.6 (6.5)	63 (113)	0.26 0.89)	--- (833 KM POLAR)	LEO

TITLE: Composite Feedlines

GENERIC CATEGORY: Thermal Performance

TECHNOLOGY ELEMENTS:

Metal liner, composite overwrap.

FLIGHT EXPERIENCE:

None.

ADVANTAGES:

Reduction of line conduction heat leak, weight savings, high strength and durability.

DISADVANTAGES:

More complex to manufacture than standard lines, technology undeveloped.

SYSTEM LEVEL DEMONSTRATIONS:

Six parallel Kapton tubes (no metal liner) were used to replace stainless lines in a cryostat (at the Laboratorium der Rijksuniversiteit te Leiden in the Netherlands), reducing heat leak into the cryostat to 64% its previous value.

DEMONSTRATED PERFORMANCE:

Straight lengths of composite lines from 1 to 22 inches in diameter utilizing a thin metallic liner overwrapped with fiberglass-epoxy were manufactured by Martin Marietta Aerospace, Denver Division. Martin Marietta also manufactured composite vacuum-jacketed lines. Both types of lines were tested for structural integrity and found to perform as designed. The vacuum-jacketed lines were also tested for vacuum retention capability and found to perform adequately.

DEMONSTRATED RELIABILITY:

No long-term reliability tests are documented. The Martin Marietta vacuum-jacketed lines were tested successfully for eight days to demonstrate vacuum retention.

PROBLEM AREAS:

Production of bends in lines, interface with pressure vessel, outgassing of composite in a vacuum environment.

KEY ISSUES:

Long-term reliability, demonstration of performance in a cryogenic system, development of non-destructive evaluation and inspection techniques.

POSSIBLE IMPROVEMENTS:

Application of low emissivity coatings to reduce radiation heat transfer, development of techniques to produce curved tube sections.

TECHNOLOGY ASSESSMENT:

Proof of concept has been performed. Technology development program required to demonstrate utilization and life time in cryogenic system.

RISK ASSESSMENT:

Composite technology outside of this application is well developed. Fully developing composite line technology would therefore entail only a moderate risk.

REFERENCES:

1. Hall, C. A., et al, "Vacuum Jacketed Composite Propulsion Feedlines for Cryogenic Launch and Space Vehicles", NASA CR134550, Martin Marietta Aerospace.
2. Thiel, R. C., et al, "Use of Kapton Film as a Cryogenic Construction Material", Cryogenics, December 1984.
3. Stark, J. A., et al, "Cryogenic Thermal Control Technology Summaries", NASA CR-134747, prepared by General Dynamics for NASA-LeRC.

TITLE: Cylindrical Heat Pipes

GENERIC CATEGORY: Thermal Performance - Heat Pipes

TECHNOLOGY ELEMENTS:

Metal envelope, wick, working fluid.

FLIGHT EXPERIENCE:

Cylindrical heat pipes have flown on numerous spacecraft. Examples: 1) 55 longitudinal grooved heat pipes were flown in May 1974 on the Applications Technology Satellite (ATS-F); 2) Three heat pipes were used to isothermalize the telescope tube on the Orbiting Astronomical Observatory (OAO-C), launched in August 1972.

ADVANTAGES:

Heat pipes are simple and reliable (no moving parts). They are capable of transporting large amounts of heat.

DISADVANTAGES:

Cylindrical heat pipes require further development in the cryogenic temperature regime.

SYSTEM LEVEL DEMONSTRATIONS:

Cylindrical heat pipes have been utilized in numerous systems (see Table A-IV for some examples).

DEMONSTRATED PERFORMANCE:

The heat transfer capability of some typical cylindrical heat pipes is shown in Table A-V.

DEMONSTRATED RELIABILITY:

Cylindrical heat pipes have been manufactured in large quantities for many years. They have been proven reliable over long periods of time. See Table A-IV for examples.

PROBLEM AREAS:

Metallic outgassing can reduce lifetime. Contamination and leakage are two other problem areas in heat pipes. However, proper manufacturing techniques can alleviate these problems.

KEY ISSUES:

More development and experience is required with heat pipes in the cryogenic temperature range.

POSSIBLE IMPROVEMENTS:

Development of production cryogenic heat pipes.

TECHNOLOGY ASSESSMENT:

Generic heat pipe technology is highly developed. Technology of cryogenic heat pipes is developing.

RISK ASSESSMENT:

Development of cryogenic heat pipes incures a medium level risk, due to contamination problems at cryogenic temperatures.

REFERENCES:

1. Kosson, R. and Grodzka, P., "Highlights in Heat Pipes and Space Processing", ATAA 75-299, presented at the AIAA 11th Annual Meeting, February 1975.
2. Fester, D. A., et al, "Long-Term Cryogenic Storage Study", AFRPL TR-82-041, prepared by Martin Marietta Aerospace for the Air Force Rocket Propulsion Lab.
3. Fester, D. A., et al, "Proceedings of the Long-Term Cryogenic Storage Technology Conference", May 12-13, 1982, at Martin Marietta Aerospace, Denver Division.

Table A-IV (Reference 3).
CYLINDRICAL HEAT PIPE PRODUCTION AND APPLICATION.

APPLICATION	PRODUCTION RATE PER YEAR	YEARS IN SERVICE
Trident Missile 108 per system	2000	10
Cruise Missile 12 per missile	6000	4
Spacecraft 8 per tube wave transformer	100	6
Rassar Array	1200	12

Table A-V. CYLINDRICAL HEAT PIPE PERFORMANCE.

TYPE	DIAMETER, cm(in)	CAPACITY, watt-cm (Btu-in/sec.)
Conventional Homogeneous Wides used for Electronic Component Cooling	1.27 (1/2)	3810 (1.42)
Grooved Wick Heat Pipes for Zero-G Applications	1.27 (1/2)	12,700 (4.74)
Centercore Wick Design	1.27 (1/2)	7620 (2.84)
Tunnel Wick Design	2.54 (1)	387,600 (144.7)

TITLE: Shadow Shields

GENERIC CATEGORY: Thermal Performance

TECHNOLOGY ELEMENTS:

Thermal control coatings, support structure, shield material.

FLIGHT EXPERIENCE:

To date, no shadow shields have been flown for use in thermal control of a cryogenic tank.

ADVANTAGES:

Shadow shields block solar flux and I. R. flux, either planetary or from a payload, thus reducing cryogen heat leak and boiloff.

DISADVANTAGES:

Shadow shields can impose orientation constraints on the storage system. They are most effective on interplanetary missions where the space vehicle can remain sun-oriented.

SYSTEM LEVEL DEMONSTRATIONS:

Several system level shadow shield tests have been performed at NASA-LeRC. Although conducted on systems of various sizes, all were similar in setup. A shadow shield was placed between a heated plate acting as a payload simulator and a cryogenic tank. The test fixtures were tested in a thermal vacuum chamber to simulate deep space conditions. The tests are described in detail in References 2, 3, 4 and 5.

DEMONSTRATED PERFORMANCE:

The NASA-LeRC tests described above demonstrated a reduction of heat leak to the cryogenic tank by a factor of 4 to 30, depending on test configuration and payload simulator temperature.

DEMONSTRATED RELIABILITY:

No shadow shield systems for protection of cryogens have been tested for long-term reliability. However, such systems are inherently reliable due to lack of moving parts and relative simplicity of design. Some long-term decrease in performance could be expected to occur due to degradation of thermal coatings.

PROBLEM AREAS:

Conduction heat leak of shield supports, degradation of optical coatings.

KEY ISSUES:

Flight demonstration of shields, demonstration of ability to shield solar flux.

POSSIBLE IMPROVEMENTS:

Use of advanced composites for conduction heat leak reduction, development of inflatable low weight sun shields.

TECHNOLOGY ASSESSMENT:

Technology is well understood, but requires further development.

RISK ASSESSMENT:

Low risk is involved in further development.

REFERENCES:

1. Boyle, R. J., and Knoll, R. H., "Thermal Analysis of Shadow Shields and Structural Members in a Vacuum", NASA TND-4876.
2. Boyle, R. J., and Stochl, R. J., "Analytic and Experimental Evaluation of Shadow Shields and their Support Member for Thermal Control of Space Vehicles", NASA TND-7612.
3. Boyle, R. J., and Stochl, R. J., "An Analytical and Experimental Evaluation of Shadow Shields and Their Support Members", Advances in Cryogenic Engineering, Volume 18, Plenum Press.

4. Boyle, R. J., et al, "Shadow Shield Experimental Studies", Proceeding of the Conference of Long-Term Cryo-Propellant Storage in Space, Marshall Space Flight Center, October 12-13, 1966.

5. Miao, D., et al, Design, "Fabrication and Structural Testing of a Lightweight Shadow Shield for Deep Space Application", NASA TND-8319.

TITLE: Dual Stage Supports

GENERIC CATEGORY: Thermal Performance

TECHNOLOGY ELEMENTS:

Fiberglass or graphite epoxy struts.

FLIGHT EXPERIENCE:

None.

ADVANTAGES:

Reduce on-orbit dewar support heat leak.

DISADVANTAGES:

More costly and complex than normal support system. Difficult to integrate and adjust in a tank to obtain proper disconnect gap.

SYSTEM LEVEL DEMONSTRATIONS:

None.

CURRENT DEVELOPMENT PROGRAMS:

1. <u>Lockheed Palo Alto - PODS III</u>

   Passive Orbital Disconnect Strut (tension-compression)

   Principle of Operation - Elastic deformation due to launch loads.

   Demonstrated Performance - Ground thermal vacuum conductance test performed with LHe sink to verify orbital conductance values.

$$G_{orbit} / G_{launch} = 6.6\%$$

$$G_{orbit} = .00008 \text{ W/K} \left(.0005 \frac{\text{Btu}}{\text{hr}^{\circ}\text{F}}\right) \qquad G_{launch} = .0012 \text{ W/K} \left(.0074 \frac{\text{Btu}}{\text{hr}^{\circ}\text{F}}\right)$$

Extensive structural loading tests have performed on the PODS III system.

2. Ball Aerospace - RITS

Rod in Tube Support - Tension support only

Principle of Operation - Elastic deformation due to launch loads

Demonstrated Performance - The RITS support has not been tested thermally or structurally. Structural and thermal design analysis has been performed. Vibration testing to determine damping pad configuration has been completed.

3. Other Conceptual Designs

Numerous conceptual designs have been mentioned in literature. They are as follows:

Differential Temperature Expansion
Mechanical Disconnect (solenoid, pyrotechnic, etc.)
Pizeolectric Expansion
Sublimation of Solid Cryogen
Magnetic
Cut Filaments
NiTi Memory Alloy

To date, none have past beyond the conceptual design phase.

DEMONSTRATED RELIABILITY:

The PODS III strut has undergone extensive structural cycling tests. These tests have led Lockheed to conclude PODS III is ready for flight applications. PODS is currently baselined for use on the SIRTF satellite.

KEY ISSUES:

System level demonstration of struts to verify predicted performance.

POSSIBLE IMPROVEMENTS:

Use of magnetic supports would eliminate on-orbit support conduction heat leak (leaving radiation leak only).

TECHNOLOGY ASSESSMENT:

Elastic deformation strut technology rapidly developing, and is nearly ready for flight applications. Other technologies would require complete development programs, but with sufficient development, could mature in the 1990 time frame.

RISK ASSESSMENT:

Use of PODS III entails relatively low risk due to extensive development. Use of other concepts are high risk due to lack of development.

REFERENCES:

1. Parmley, R. T. and Kittel, P., "Passive Orbital Disconnect Strut (PODS III)", Advances in Cryogenic Engineering, Volume 29, Plenum Press.
2. Parmley, R. T. and Kittel, P., "System Structural Test Results: Six PODS III Supports", presented at the 1985 Cryogenic Engineering Conference.
3. Hopkins, D. A., "Structural Support Release Systems for Cryogenic Coolers", Ball Aerospace Report F83-06, Contract NAS5-27247, performed for NASA Goddard Space Flight Center.
4. Fester, D. A., "Long-Term Cryogenic Storage Study", AFPRL-TR-82-071, performed by Martin Marietta Aerospace for the Air Force Rocket Propulsion Laboratory.
5. Parmley, R. T., et al, "Test and Evaluate Passive Orbital Disconnect Struts (PODS-III)", NASA CR 177368, August 1985, Lockheed Missiles and Space Company.

TITLE: Thermal Control Coatings

GENERIC CATEGORY: Thermal Performance

TECHNOLOGY ELEMENTS:

Films, deposited coatings, paint, fabrics, tapes.

FLIGHT EXPERIENCE:

Thermal control coatings have been used on virtually all space hardware flown Example: 1) Space Shuttle cargo bay is lined with beta cloth $\alpha/\epsilon = 0.32/0.86$; 2) manned maneuvering unit white paint $\alpha/\epsilon = 0.3/0.85$.

ADVANTAGES:

Thermal control coatings are used to maintain spacecraft temperatures within acceptable limits. They are reliable and simple compared to active systems.

DISADVANTAGES:

Thermal control coatings undergo degradation when exposed long-term in an orbital environment. Thermal control films can experience debonding due to thermal cycling.

SYSTEM LEVEL DEMONSTRATIONS:

Thermal control coatings are used in virtually every spacecraft thermal control system.

DEMONSTRATED PERFORMANCE:

Depending on desired temperatures, heat flux and configuration, a coating with desired α/ϵ characteristics is chosen. See Figure A-2 for some representative coatings and their effect on equilibrium temperature.

DEMONSTRATED RELIABILITY:

Thermal control coatings are reliable, but experience degradation with time. Table A-VI and Figure A-3 show examples of this degradation.

PROBLEM AREAS:

Long-term stability of optical properties, atomic oxygen degradation.

KEY ISSUES:

Synergistic effect of coating contamination and orbital environment. Static charge buildup in geo-synchronous orbits.

POSSIBLE IMPROVEMENTS:

Improve long-term life of coatings and determination of which coatings undergo the least degradation. Data from the Long Duration Exposure Facility (LDEF) will aid in determining which coatings have the longest life.

TECHNOLOGY ASSESSMENT:

The technology of producing and applying coatings with virtually any α/ϵ is well defined. Table A-VII and Figure A-4 show some representative samples of a few of the available coatings.

RISK ASSESSMENT:

Technology is well understood and incures minimal risk in utilization.

REFERENCES:

1. Conan, S. M. and Chow, D. T., "Thermal Control Film Evaluation for Space Applications", presented at the 11th National SAMPE Technical Conference, November 13-15, 1974.
2. Kay, A., "SIRTF Thermal Control Surfaces", Beech Aircraft MR-17228.
3. Fester, D. A., et al, "Long-Term Cryogenic Storage Study", AFRPL TR-82-071, prepared by Martin Marietta Aerospace for the Air Force Rocket Propulsion Lab.

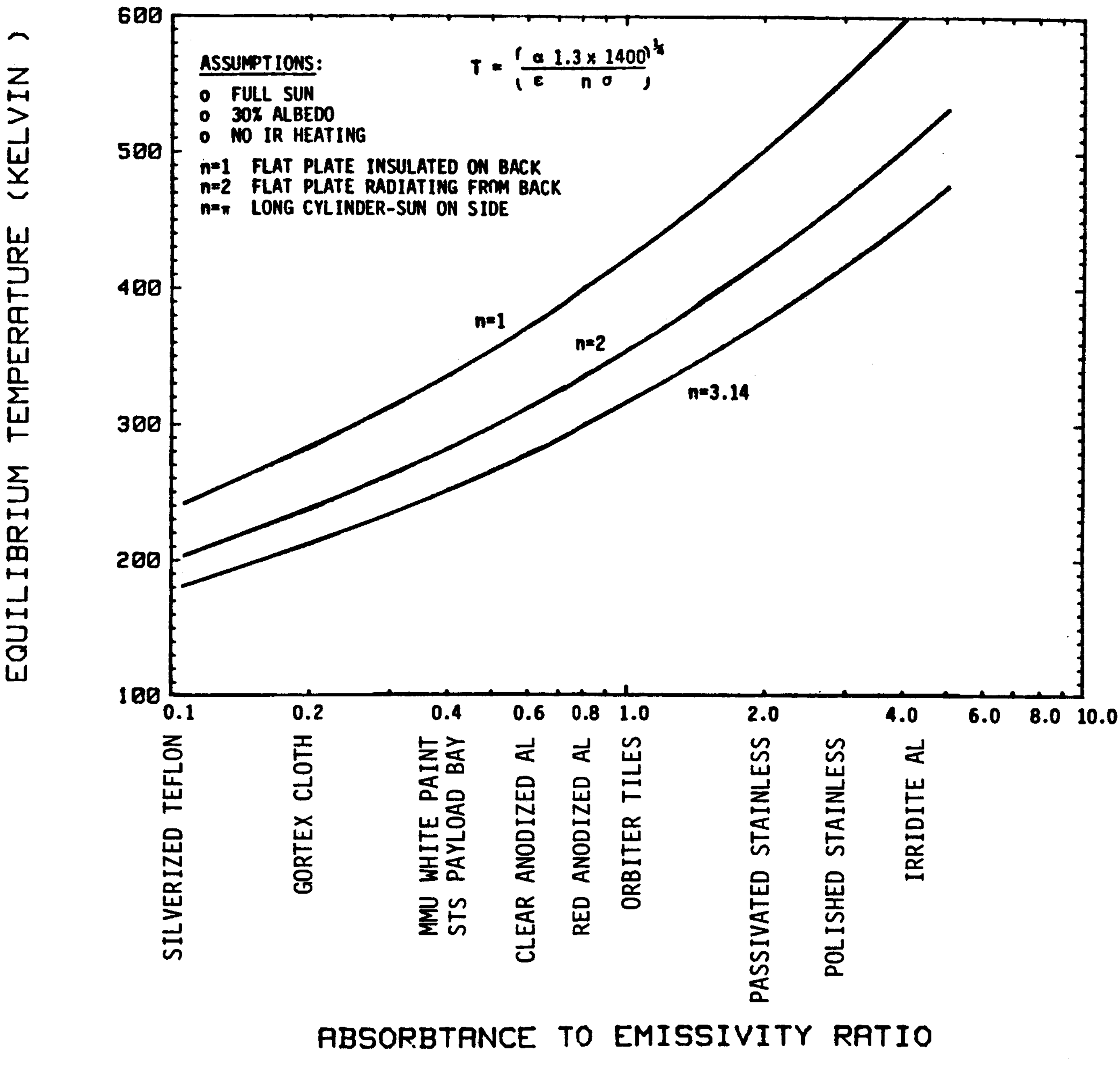
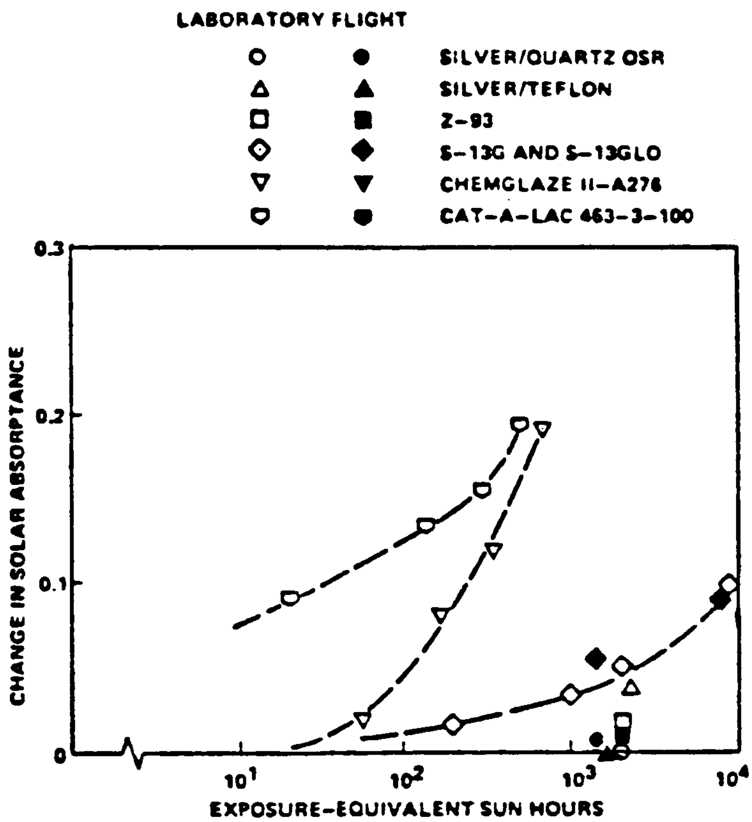


Figure A-2. EQUILIBRIUM TEMPERATURE VS α/ϵ .

Table A-VI (Reference 3).
THERMAL CONTROL COATINGS DEGRADATION.

COATING	INITIAL α/ϵ	ORBIT	DEGRADED α/ϵ	TIME IN ORBIT
AG TEFLON	.13	LEO	.14	1.5 YR
AG TEFLON	.20	LEO	.22	1.5 YR
AL & SIO_x	.50	GEO	.66	1 YR
AL & SIO_x	.50	LEO	.51	1.5 YR
AL & AL_2O_3	.59	GEO	.66	1 YR
MS-74 PAINT	.23	LEO	.24	1.5 YR
MS-74 PAINT	.23	GEO	.47	1 YR



REF. R. SCHWINGHAMER, "SPACE ENVIRONMENTAL EFFECTS ON MATERIALS", NASA TM-78306, AUGUST 1980.

Figure A-3. STABILITY OF THERMAL CONTROL COATINGS IN UV/VACUUM EXPOSURE (LEO)

Table A-VII (Reference 2).
NOMINAL ROOM-TEMPERATURE, THERMO-OPTICAL CHARACTERISTICS OF THERMAL-CONTROL MATERIALS.

MATERIAL	DESCRIPTION	MANUFACTURER	SOLAR ABSORPTANCE (α_s)	INFRARED EMITTANCE (ϵ)
White Paints:				
Kemacryl	TiO_2/Acrylic	Sherwin-Williams Co.	0.28	0.86
Skyspar	TiO_2/Epoxy	Andrew Brown Co.	0.22	0.86
DC 92-007 (Thermatrol)	TiO_2/Silicone	Dow Corning Corp.	0.17	0.84
S-13	ZnO/Silicone	Illinois Inst. of Technology Institute (IITRI)	0.20	0.87
S-13G	ZnO/Silicone	IITRI	0.20	0.85
Z-93	ZnO/K_2SiO_3	IITRI	0.15	0.87
LP-10A	$ZrSiO_4/K_2SiO_3$	Lockheed Missiles and Space Co. (LMSC)	0.12	0.86
Tapes:				
Mystic's 7-102L	Aluminum foil with silicone adhesive	Mystic Tape Co., Div of Borden Chemical	0.14	0.03
Permacel's EE-6600	Aluminized polyester with rubber adhesive	Permacel Tape Co.	0.12	0.03
3M's 850	Polyester, aluminized on backside with acrylic adhesive	3M Co.	0.15	0.61
3M's Y-9184	Goldized Polyimide with acrylic adhesive	3M Co.	0.24	0.03
Films:				
Aluminized Polyester	See Material column	National Metallizing, Div. of Standard Packaging Corp.	0.12	0.03
Aluminized Polyimide	See Material column	National Metallizing, Div. of Standard Packaging Corp.	0.12	0.03
Second Surface Mirrors:				
Optical Solar Reflector (OSR)	.008 inch-thick quartz with vacuum-deposited silver and Inconel on one side	Optical Coatings Laboratory, Inc.	0.05	0.80
Aluminized Teflon	.005 inch-thick Type A FEP teflon with aluminum deposited on one side	G. T. Schjeldahl Companpy Product No. G-400900	0.12	0.85
Silvered Teflon	.005 inch-thick Type A FEP teflon with silver and Inconel on one side	G. T. Schjeldahl Company Product No. G-401500	0.07	0.85
Dielectric-Overcoated Aluminized Polyimide	.0005 inch-thick polyimide film with topcoating of vapor-deposited aluminum and silicon oxide	G. T. Schjeldahl Company Product No. G-101500	0.15	0.40
Special Finishes & Materials:				
Clear Sulfuric-Acid Anodized Aluminum	Electrodeposited anodic coating on aluminum	MIL-A-8625, Type II, Class I, Clear Seal	0.19-0.30	0.76
Black Sulfuric-Acid Anodized Aluminum	Dyed, electrodeposited anodic coating on aluminum	MIL-A-8625, Type II, Class II, Black Dyed	0.65	0.80
Chromate-Anodized Magnesium	Chromated, electro-deposited coating on magnesium	MIL-M-45202, Type I, Class C	0.53-0.72	0.71-0.82
Gold Plating	Electrodeposited gold coating	MIL-G-45204, Type I, Class 5	0.26	0.03
Clad 2024 Aluminum	Structural aluminum alloy	Alcoa, Kaiser, Reynolds, etc.	0.14-0.28	0.04
Clad 7075 Aluminum	Structural aluminum alloy	Alcoa, Kaiser, Reynolds, etc.	0.19-0.33	0.04
Anodized Titanium Foil	.0005 inch-thick titanium foil anodized in tartaric acid bath	Lockheed Missiles & Space Co.	0.70	0.10
Barrier Anodized Aluminum	Clear anodize on specular, polished, aluminum substrate	The Boeing Company	0.12-0.30	0.07-0.40

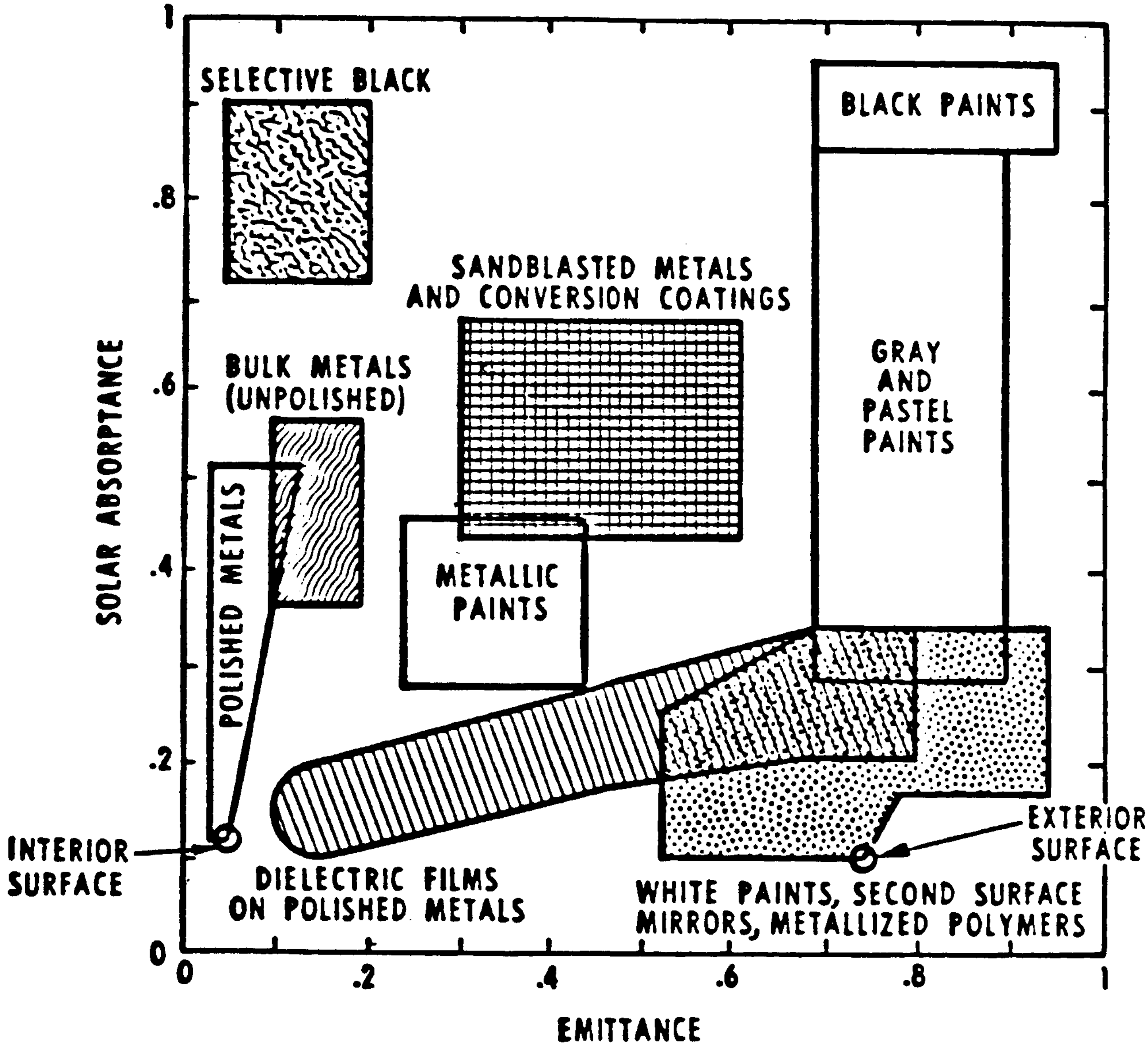


Figure A-4 (Reference 2).
COATINGS MAP.

TITLE: Thermodynamic Vent Systems

GENERIC CATEGORY: Thermal Performance

TECHNOLOGY ELEMENTS:

Heat exchanger, Joule-Thomson valves, vapor cooled shields.

FLIGHT EXPERIENCE:

Cryogenic tanks utilizing vapor cooled shields were flown on Apollo (H_2 and O_2 fuel cell tanks), the Infra-Red Astronomical Satellite (SfHe), and are currently being flown on the Space Shuttle (H_2 PRSA tank). See Table A-VIII for a listing of PRSA flight experience.

ADVANTAGES:

Thermodynamic Vent Systems (TVS) reduce heat flux and boil-off in a cryogenic vessel. For two-phase systems, with positive Joule-Thomson coefficients, use of a Joule-Thomson valve allows efficient use of the cooling available in any liquid that may be vented.

DISADVANTAGES:

Added system weight, increased complexity and manufacturing costs.

SYSTEM LEVEL DEMONSTRATIONS:

System level flight demonstrations have been performed as indicated above. Martin Marietta has demonstrated the concept using three one-g test articles ranging from a 0.33m (1.1 ft.) diameter sphere to a 1.2m (3.9 ft) diameter by 1.8m (5.9 ft) long tank. Martin Marietta has also performed a ground demonstration of a coupled H_2/N_2 thermodynamic vent system. In this system, vented hydrogen was used to cool both the H_2 and N_2 tanks. Due to the higher temperature of the N_2, it was maintained in a no-vent condition. Beech Aircraft has performed several ground tests of dewars utilizing vapor cooled shields, in addition to the Apollo and Shuttle flight articles listed above. Further data on these tests are listed in Table A-X. General Dynamics Convair built and demonstrated a LOX TVS with Joule-Thomson valve and internal heat exchanger in 1974 (Reference 5).

DEMONSTRATED PERFORMANCE:

The improvement in performance for hydrogen dewars through use of vapor cooled shields is shown in Table A-IX for various tanks during ground test. The theoretical improvement in performance for a hydrogen dewar through use of one to four vapor cooled shields is depicted in Figure A-5.

DEMONSTRATED RELIABILITY:

Thermodynamic vent systems are inherently reliable due to lack of moving parts. The IRAS vapor cooled shields operated reliably on-orbit for a period of 10 months. The vapor cooled shields on the Apollo fuel cell tanks operated reliably during 11 flights. The Space Shuttle PRSA tanks have accumulated a total flight time of 3634 hours without failure (see Table A-VIII).

PROBLEM AREAS:

Control of complex thermodynamic vent systems (multiple heat exchangers, coupled systems, etc), contamination of Joule-Thomson valves.

KEY ISSUES:

Optimization and control of complex thermodynamic vent systems, demonstration of coupled H_2/O_2 systems. Demonstration of Para-Ortho hydrogen conversion in a thermodynamic vent system.

POSSIBLE IMPROVEMENTS:

Use of Para-Ortho hydrogen conversion to increase cooling capacity. Development of long-lifetime microprocessor controller for optimal control of complex thermodynamic vent systems utilizing Joule-Thomson valves, internal heat exchangers, vapor cooled shields, and convective cooling of supports and penetrations.

TECHNOLOGY ASSESSMENT:

The technology of vapor cooled shields is well developed and shown flightworthy. Further development and demonstration required for more complex thermodynamic vent systems.

RISK ASSESSMENT:

There is minimal risk involved in further development of thermodynamic vent systems. Resolution of key issues present no major technological obstacles.

REFERENCES:

1. Fester, D. A., "Long-Term Cryogenic Storage Study", AFRPL TR-82-071, prepared by Martin Marietta Aerospace for the Air Force Rocket Propulsion Lab.
2. Hopkins, R. A., "Design of a One-Year Lifetime, Spaceborne Superfluid Helium Dewar", Ball Aerospace Systems Division, Boulder, Colorado.
3. Gier, H. L., "Tank Summary", Beech Aircraft, MR-14708A.
4. Maiden, T. E., "Cryogenic Tank Support and Insulation Summary", Beech Aircraft, MR-14933.
5. Erickson, R. C., "Space LOX Vent System", CASD-NAS75-021, April 1975, prepared by General Dynamics Convair for NASA Marshall Space Flight Center.

Table A-VIII. SPACE SHUTTLE - PRSA FLIGHT TIME LOG

NO.	FLIGHT NO.	VEHICLE	FLIGHT HOURS	TOTAL HOURS	DATE
1	STS-1	102	54.0	54.0	
2	STS-2	102	54.2	108.2	
3	STS-3	102	192.3	300.5	
4	STS-4	102	169.1	469.6	
5	STS-5	102	122.2	591.8	
6	STS-6	099	121.0	712.8	
7	STS-7	099	146.0	858.8	06/18/83
8	STS-8	099	145.0	1003.8	08/30/83
9	STS-9	102	240.0	1243.8	11/28/83
10	STS-11	099	191.3	1435.1	02/03/84
11	41C	099	167.5	1602.6	04/06/84
12	41D	103	145.0	1747.6	08/30/84
13	41G	099	199.4	1947.0	10/05/84
14	51A	103	192.0	2139.0	11/08/84
15	51C	103	73.5	2212.5	01/24/85
16	--	103	168.5	2381.0	04/12/85
17	51B	099	168.0	2549.0	04/24/85
18	51G	103	169.0	2718.0	06/17/85
19	51F	099	168.8	2886.8	07/29/85
20	51I	103	170.2	3057.0	08/27/85
21	51J	104	97.8	3154.8	10/03/85
22	61A	099	168.8	3323.6	10/30/85
23	61B	104	165.0	3488.6	11/26/85
24	61C	102	146.0	3634.6	01/12/86

Table A-IX. HEAT LEAK IMPROVEMENT THROUGH USE OF VAPOR COOLED SHIELDS.

	BEECH AIRCRAFT HTTA	BEECH AIRCRAFT PRSA - H_2	BEECH AIRCRAFT ELMS
Tank Volume	22.7 m^3 (800 ft^3)	.606 m^3 (21.4 ft^3)	1.27 m^3 (45 ft^3)
Cryogen	H_2	H_2	H_2
Heat Leak No VCS Flow	5.01 watts (17.1 Btu/hr)	7.41 watts (25.3 Btu/hr)	9.67 watts (33 Btu/hr)
Heat Leak With VCS Flow	1.88 watts (6.4 Btu/hr)	2.75 watts (9.4 Btu/hr)	1.46 watts (5 Btu/hr)

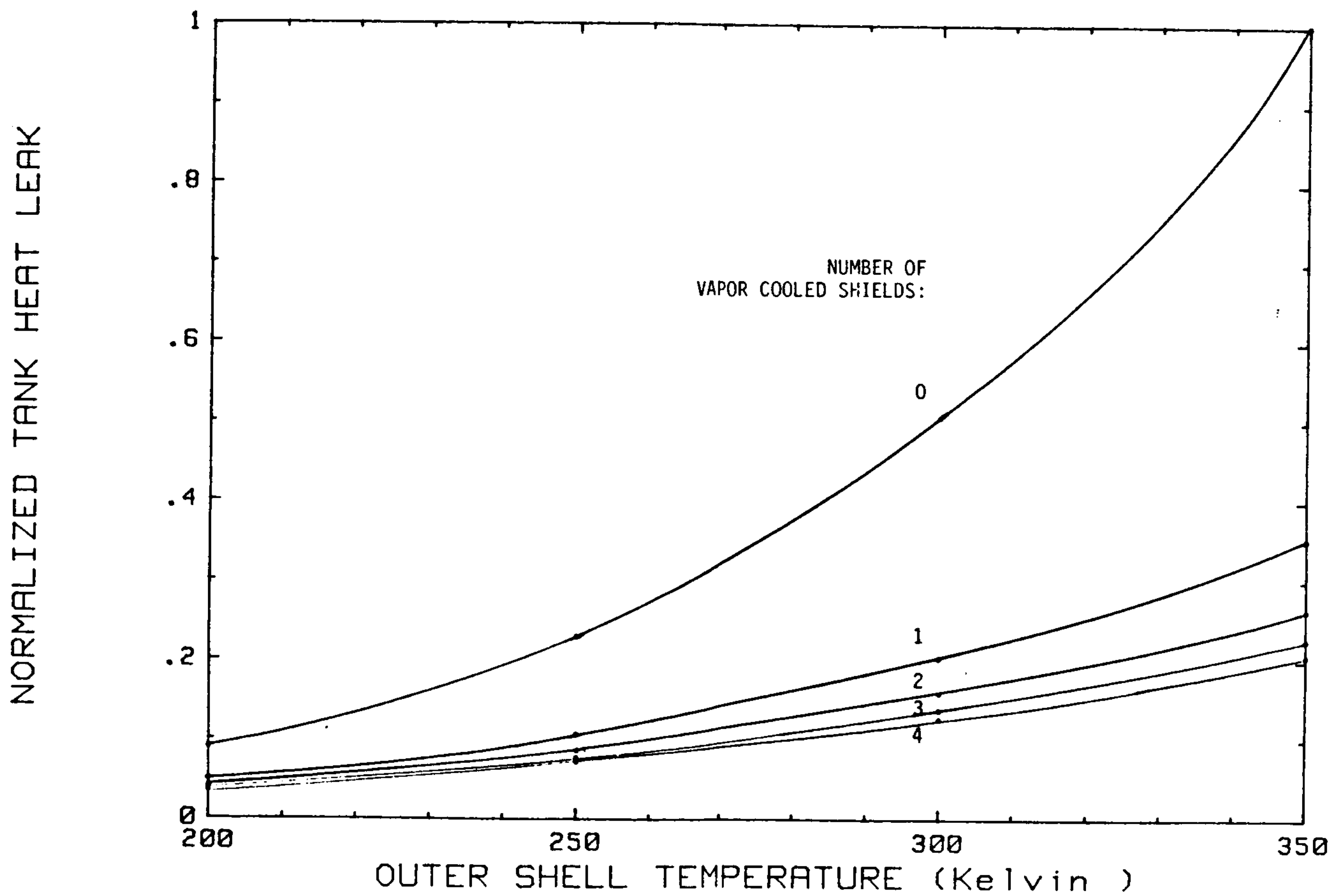


Figure A-5. H_2 DEWAR PERFORMANCE IMPROVEMENT THROUGH USE OF VAPOR COOLED SHIELDS.

TITLE: Thermal Stratification Control

GENERIC CATEGORY: Thermal Performance

TECHNOLOGY ELEMENTS:

Mixing fans, tank mounted heat exchangers, mixing jets.

FLIGHT EXPERIENCE:

Mixing fans were utilized in the Apollo H_2 and O_2 fuel cell tanks for stratification control.

ADVANTAGES:

Stratification control reduces vent losses and pressure transients in tanks when destratification due to sudden mixing occurs. By maintaining the cryogen more nearly in equilibrium, more accurate quantity gaging, temperature and pressure sensing is possible.

DISADVANTAGES:

Stratification control devices add to the cost and complexity of a dewar. Mixing fans require power for operation, increasing dewar heat input, which is undesirable for long-term storage. Safety is an issue in usage with oxygen tanks.

SYSTEM LEVEL DEMONSTRATIONS:

The Apollo Fuel Cell tanks demonstrated zero-g stratification control on a system level for supercritical storage. General Dynamics Convair built and ground-demonstrated a LOX internal heat exchanger (Reference 4).

DEMONSTRATED PERFORMANCE:

The Apollo O_2 Fuel Cell tank would experience a 689 kPa (100 psid) pressure drop during booster separation if the mixing fans were not operated. Operation of the mixing fans prior to separation reduced this drop to 345 kPa (50 psid).

DEMONSTRATED RELIABILITY:

No stratification devices have demonstrated long-term reliable operation. The Apollo devices experienced one failure, attributable to pre-flight handling rather than unreliability of device.

PROBLEM AREAS:

Heat input from active mixing devices.

KEY ISSUES:

Development and demonstration of devices suitable for long-term storage. Further analysis and technological development required in understanding impacts of stratification and key issues pertaining to long-term cryogenic storage.

POSSIBLE IMPROVEMENTS:

Utilization of heat pipes to provide uniform fluid state. Utilization of thermodynamic vent system tank heat exchangers to control stratification.

TECHNOLOGY ASSESSMENT:

Technology is moderately developed in certain areas (active mixers). Technology utilizing tank wall heat exchangers is developing out of the Cryogenic Fluid Management Flight Experiment (CFMFE) Program.

RISK ASSESSMENT:

Further development entails moderate risk. Further understanding of stratification issues in relationship to long-term storage is necessary to evaluate costs vs. benefits of this technology.

REFERENCES:

1. Fester, D. A., et al, "Long-Term Cryogenic Storage Study", AFRPL TR-82-071, prepared by Martin Marietta Aerospace for the Air Force Rocket Propulsion Lab.
2. Lester, J. M. and Hickman, W. H., "Zero-Gravity Thermal Performance of the Apollo Cryogenic Gas Storage System", Advances in Cryogenic Engineering, Volume 16, 1970, Plenum Press.
3. Stark, J. A., et al, "Fluid Management Systems Technology Summary", NASA CR-134748.
4. Erickson, R. C., "Space LOX Vent System", CAD S-NAS75-021, April 1975, prepared by General Dynamics Convair for NASA Marshall Space Flight Center.

TITLE: Thick MLI

GENERIC CATEGORY: Thermal Performance

TECHNOLOGY ELEMENTS:

Low emissivity metallic coated mylar or Kapton, low thermal conductance spacer material

FLIGHT EXPERIENCE:

Tanks with MLI insulation systems have extensive flight experience. Examples include: Gemini, Apollo and Space Shuttle (PRSA) fuel cell tanks, and the Infrared Astronomical Satellite (IRAS). Although only IRAS could be considered having thick MLI, the use of these insulation systems has been shown to be flight-worthy.

ADVANTAGES:

Lightweight, lowest heat leak of any type of cryogenic insulation system.

DISADVANTAGES:

Multilayer insulation is very labor-intensive to install. Thick MLI requires long pumpdown times.

SYSTEM LEVEL DEMONSTRATIONS:

Thick MLI systems have performed in space (IRAS) and in numerous ground tests of high performance dewars. See Table A-X for examples.

DEMONSTRATED PERFORMANCE:

See Table A-X for performance figures on several high performance hydrogen dewars that have been ground tested.

DEMONSTRATED RELIABILITY:

Multilayer insulation systems are passive, and thus inherently reliable. No system performance and reliability data are available for periods greater than one year. However, the IRAS dewar insulation system performed satisfactorily for 10 months on-orbit and the Shuttle PRSA tanks have been reflown numerous times.

PROBLEM AREAS:

Degradation of MLI optical properties, performance degradation due to penetrations, labor-intensive buildup. Effects of vibration and static loads on thick blankets.

KEY ISSUES:

Development of efficient lower cost insulation layup techniques. Evacuation of very thick insulation systems. Structural support of thick systems to survive STS launch loads.

POSSIBLE IMPROVEMENTS:

Development of more efficient layup techniques to decrease manufacturing costs.

TECHNOLOGY ASSESSMENT:

The technology of MLI is well defined. Refinement is needed in area of thick MLI layup techniques.

RISK ASSESSMENT:

There are no significant unknowns involved in further development. There is very low risk involved in further development.

REFERENCES:

1. Fester, D. A., et al, "Proceedings of the Long-Term Cryogenic Technology Conference", May 12-13, 1982, at Martin Marietta Aerospace, Denver Division.
2. Fester, D. A., et al, "Long-Term Cryogenic Storage Study", AFRPL TR-82-041, prepared by Martin Marietta Aerospace for the Air Force Rocket Propulsion Laboratory.
3. Gier, H. L., "Tank Summary", Beech Aircraftr, MR-14708A.
4. Maiden, T. E., "Cryogenic Tank Support and Insulation Summary", Beech Aircraft, MR-14933.
5. Stark, J. A., et al, "Cryogenic Thermal Control Technology Summaries", NASA CR-134747, prepared by General Dynamics for NASA-Lewis Research Ctr.

Table A-X. HYDROGEN TANK PERFORMANCE.

	BEECH AIRCRAFT 1972	GENERAL DYNAMICS 1969	MCDONNEL DOUGLAS 1973	NASA-LEWIS 1977	NASA-LEWIS 1978	NASA-MSFC 1972
No. of Supports	22 Straps	6 Struts	1 Strut	12 Struts	6 Struts	1 Strut
Radiation Shield	DSM	DGK	DAM	DAM	DAM	DGK
Spacer Material	Nylon/Silk	Dacron Tufts	Dacron	2-Silk	2-Silk	Dacron Tufts
Layer Density, Layers/cm (layers/in) Total No. Layers	8 (20) 54	11 (29) 44	16 (40) 28	18 (45) 34	18 (45) 34	12 (30) 60
Tank Volume, m^3 (ft^3)	22.7 (800)	5.66 (200)	10.3 (364)	5.66 (200)	1.4 (51)	12.7 (450)
Heat Flux, W/m^2	0.107 0.041*	0.826	0.318	1.47	1.44	0.234
($Btu/hr\text{-}ft^2$)	(0.034) (0.013)*	(0.262)	(0.101)	(0.468)	(0.458)	(0.0742)

* WITH VAPOR COOLED SHIELDS IN OPERATION

TITLE: Capillary Acquisition

GENERIC CATEGORY: Fluid Management Transfer Systems

TECHNOLOGY ELEMENTS:

Start baskets, start tanks, acquisition liners and channels.

FLIGHT EXPERIENCE:

Space Shuttle Orbital Maneuvering System (OMS) and Reaction Control System (RCS) tanks, Viking Orbiters (2 flights), RCA SATCOM Satellite, Agena upper stage (120 flights), Apollo Service Module (11 flights). All of these systems were for use with earth storable fluids.

ADVANTAGES:

Allows acquisition of subcritical fluids in zero-g, eliminates need for settling thrusters.

DISADVANTAGES:

Application to cryogenic tanks needs technical development, increased weight, cost and complexity of tanks.

SYSTEM LEVEL DEMONSTRATIONS:

Capillary acquisition devices have performed successfully on numerous spacecraft using earth storable fluids (see Flight Experience).

DEMONSTRATED PERFORMANCE:

Viking vane-type acquisition device achieved nearly 100% expulsion efficiency over a 5-year period, typical acquisition devices have expulsion efficiencies over 96%.

DEMONSTRATED RELIABILITY:

Capillary acquisition devices are passive and have performed reliably for long-term periods of time. A capillary acquisition device performed reliably for 5 years on the Viking orbiters.

PROBLEM AREAS:

Boiling of cryogens in acquisition devices.

KEY ISSUES:

Development of technology for cryogenic fluids. Virtually all capillary devices used to date were used with earth-storable fluids, such as hydrazine.

POSSIBLE IMPROVEMENTS:

Development of lighter weight, simpler to manufacture, acquisition devices.

TECHNOLOGY ASSESSMENT:

Technology is well developed and flight proven, but needs further development with cryogenic fluids.

RISK ASSESSMENT:

Application to long-term cryogenic storage entails a medium level risk.

REFERENCES:

1. DeBrock, S. C., "Development and Flight Experience with a Capillary Propellant Management System for a Three Axis Stabilized Vehicle", Lockheed Missiles and Space Company.
2. DeBrock, S. C., "Spacecraft Capillary Propellant Retention and Control for Long-Life Missions", Lockheed Missiles and Space Company.
3. Sloma, R. O., "Capillary Propellant Management System for Large Tank Orbital Propulsion Systems", Lockheed Missiles and Space Company.
4. Dominick, S. M. and Tegart, J. R., "Low-G Propellant Transfer using Capillary Devices", AIAA-81-1507, presented at the AIAA/SAE/ASME Joint Propulsion Conference, July 28-29, 1981.
5. Blatt, M. H., et al, "Capillary Acquisition Devices for High-Performance Vehicles", Executive Summary, NASA CR-159658, prepared by General Dynamics for NASA-LeRC.
6. Stark, J. A., "Fluid Management Systems Technology Summaries", NASA CR-134748, prepared by General Dynamics for NASA-LeRC.

TITLE: Honeycomb Composite Outer Shells

GENERIC CATEGORY: Weight Reduction

TECHNOLOGY ELEMENTS:

Aluminum honeycomb core, fiberglass or aluminum face sheets, permeability layer.

FLIGHT EXPERIENCE:

None.

ADVANTAGES:

Reduction of dewar mass, relatively inexpensive, short lead times.

DISADVANTAGES:

Technology not fully developed.

SYSTEM LEVEL DEMONSTRATIONS:

Two system level demonstrations of honeycomb outer shells have been performed. Boeing Aerospace manufactured a honeycomb outer shell with aluminum face sheets. The outer shell was used on a 7.25 m^3 (256 ft^3) H_2 tank, a half-scale prototype for the Shuttle OMS system. Beech Aircraft manufactured ten honeycomb outer shell hemispheres with fiberglass face sheets in various configurations. One set of outer shell hemispheres was installed on a Space Shuttle Power Reactant Supply Assembly (PRSA) H_2 tank for ground testing.

DEMONSTRATED PERFORMANCE:

The Boeing outer shell was pressure cycled from ambient to vacuum for 29 cycles, and accumulated approximately 1500 hours at vacuum pressure. The outer shell subsequently experienced a catastrophic structural failure while under vacuum due to adhesive de-bonding. Eight of the ten Beech outer shell hemispheres were collapse tested with failure occuring between 124 and 427 kPa (18 and 62 psia). These tests sucessfully demonstrated structural integrity and analytical design techniques. The widely varying collapse pressure was due to different types of core material used.

The configuration that was installed on the PRSA ground test tank had a collapse pressure of 255 kPa (37 psia). Vacuum was not successfully obtained due to leakage through the gore seams of the permeation barrier on the inner face sheet. Subsequent testing on an improved bonding method was performed on face sheet samples and found to perform adequately, but a system level test was never performed. The Beech honeycomb outer shells provided a weight savings of 50% over standard aluminum construction.

DEMONSTRATED RELIABILITY:

No long-term testing has been performed to date. Collapse pressure tests have demonstrated consistent ability to withstand design loads.

PROBLEM AREAS:

Verification of long-term vacuum retention capability.

KEY ISSUES:

Verification of vacuum retention capability on full scale article. Testing for long-term reliability, both structurally and in long-term vacuum retention. Outer shell to girth ring attachment and removal techniques need to be developed.

POSSIBLE IMPROVEMENTS:

Development and testing of honeycomb girth ring would increase potential weight savings.

TECHNOLOGY ASSESSMENT:

Proof of concept has been performed adequately. Long-term testing required before technology can be applied to flight articles.

RISK ASSESSMENT:

Technology is relatively straightforward with few anticipated problems. Development of flight articles would entail only a small risk factor.

REFERENCES:

1. Scarlotti, R. D., "Development of Honeycomb Sandwich Materials in the Construction of Cryogenic Dewar Outer Shells", Advances in Cryogenic Engineering, Volume 31, Plenum Press.
2. Barclay, D. L., et al, "Lightweight Vacuum Jacket for Cryogenic Insulation", NASA CR134759, Boeing Aerospace.

TITLE: Soft Outer Shells

GENERIC CATEGORY: Weight Reduction

TECHNOLOGY ELEMENTS:

Purge bag, purge pins, ground support gaseous purge system

FLIGHT EXPERIENCE:

Soft outer shell tanks are utilized in the Centaur upper stage vehicle.

ADVANTAGES:

Reduces system mass.

DISADVANTAGES:

Increases ground hold heat leak by several orders of magnitude, decreased on-orbit thermal performance, more sensitive to on-orbit contamination and micro-penetration.

SYSTEM LEVEL DEMONSTRATIONS:

Goodyear Corporation produced soft outer shells for a 1.78m (70 in.) and 2.67m (105 in.) diameter tank which were subsequently tested at Marshall Space Flight Center. An insulation system on a 1.4m (55 in.) diameter H_2 tank in vacuum and one atmosphere GHe was tested by NASA/LeRC without a purge bag. MSFC is testing a soft outer shell built by General Dynamics on an 2.21m (87 in.) diameter H_2 tank.

DEMONSTRATED PERFORMANCE:

The NASA LeRC test demonstrated a 1.36 W/m^2 (0.43 $BTU/hr\text{-}ft^2$) ground hold heat leak (31.4% boiloff per hour). Predicted on-orbit performance of General Dynamics tank is 0.63 W/m^2 (0.2 $BTU/hr\text{-}ft^2$).

DEMONSTRATED RELIABILITY:

To date, no soft outer shells have been tested for long-term reliability.

PROBLEM AREAS:

Ground hold heat leak, susceptability of insulation system to micrometeroid and atomic oxygen damage, lower on-orbit thermal performance due to perforations in MLI needed for ground hold purge.

KEY ISSUES:

Reducing ground hold heat leak and/or resupplying cryogen during ground hold, demonstration of long-term thermal performance.

POSSIBLE IMPROVEMENTS:

Improve ground hold thermal performance, development of outer shell with long on-orbit lifetime.

TECHNOLOGY ASSESSMENT:

Proof of concept has been performed, technology is moderately developed.

RISK ASSESSMENT:

Further development of technology for long-term storage entails a moderate risk.

REFERENCES:

1. "Soft Outer Shell Study", Beech Aircraft MR-17814, December 1982.
2. "Flexible Vacuum Jacket Development Final Report", GER13342, prepared by Goodyear Aerospace Corporation for NASA-MSFC, June 1967.
3. Conder, R. L., "Flexible Vacuum Jackets Technology Survey", Beech Aircraft MR-14900, November 1979.
4. Fester, D. A., et al, "Long Term Cryogenic Storage Study", AFRPL TR-83-082, December 1984, prepared by Martin Marietta Aerospace for the Air Force Rocket Propulsion Laboratory.

APPENDIX B

FUTURE CRYOGENIC DEVELOPMENT PROGRAMS

FUTURE CRYOGENIC DEVELOPMENT

PROGRAM TITLE: Zero-G Quantity Gaging

CRYOGENIC TECHNOLOGY: Quantity Gaging

SPONSORING AGENT: NASA - Johnson Space Center

PROGRAM OBJECTIVE: Develop and evaluate zero-gravity quantity gaging system concepts having one percent or better accuracy for on-orbit, two-phase cryogenic tankage. The immediate application is to accelerate technology suitable for providing zero-g quantity gaging for the CFMFE.

EXPECTED CRYOGENIC DEVELOPMENT: The current program underway at Beech Aircraft is Phase I of a three phase program. Phase I will evaluate technologies, then develop and test a zero-g quantity gaging system that can be utilized for two-phase cryogenic storage. Phase II will develop and test a prototype CFMFE quantity gaging system. Phase III will be a full scale development and flight qualification program, with the end result being a flight unit for use on the CFMFE program. It is expected by the early 1990s that this flight-qualified unit will be operational and suitable for application on CFMFE and other programs as needed.

FUTURE CRYOGENIC DEVELOPMENT

PROGRAM TITLE: Cryogenic Fluid Management Flight Experiment

CRYOGENIC TECHNOLOGY: Zero-G Fluid Management

SPONSORING AGENT: NASA-Lewis Research Center

PROGRAM OBJECTIVE: Develop and test on-orbit cryogenic fluid storage and transfer technologies, including capillary acquisition, quantity gaging and thermodynamic vents.

EXPECTED CRYOGENIC DEVELOPMENT: A liquid hydrogen storage and transfer experiment will be flown in the cargo bay of the Shuttle Orbiter in the early 1990s (see Figure B-1). Multiple Shuttle flights employing alternate hardware will provide the data necessary to provide fluid management system design criteria for a variety of in-space applications. The expected technology development from this experiment will be the development of a flight-proven cryogenic Liquid Acquisition Device, Thermodynamic Vent System and Low-g Quantity Gaging System.

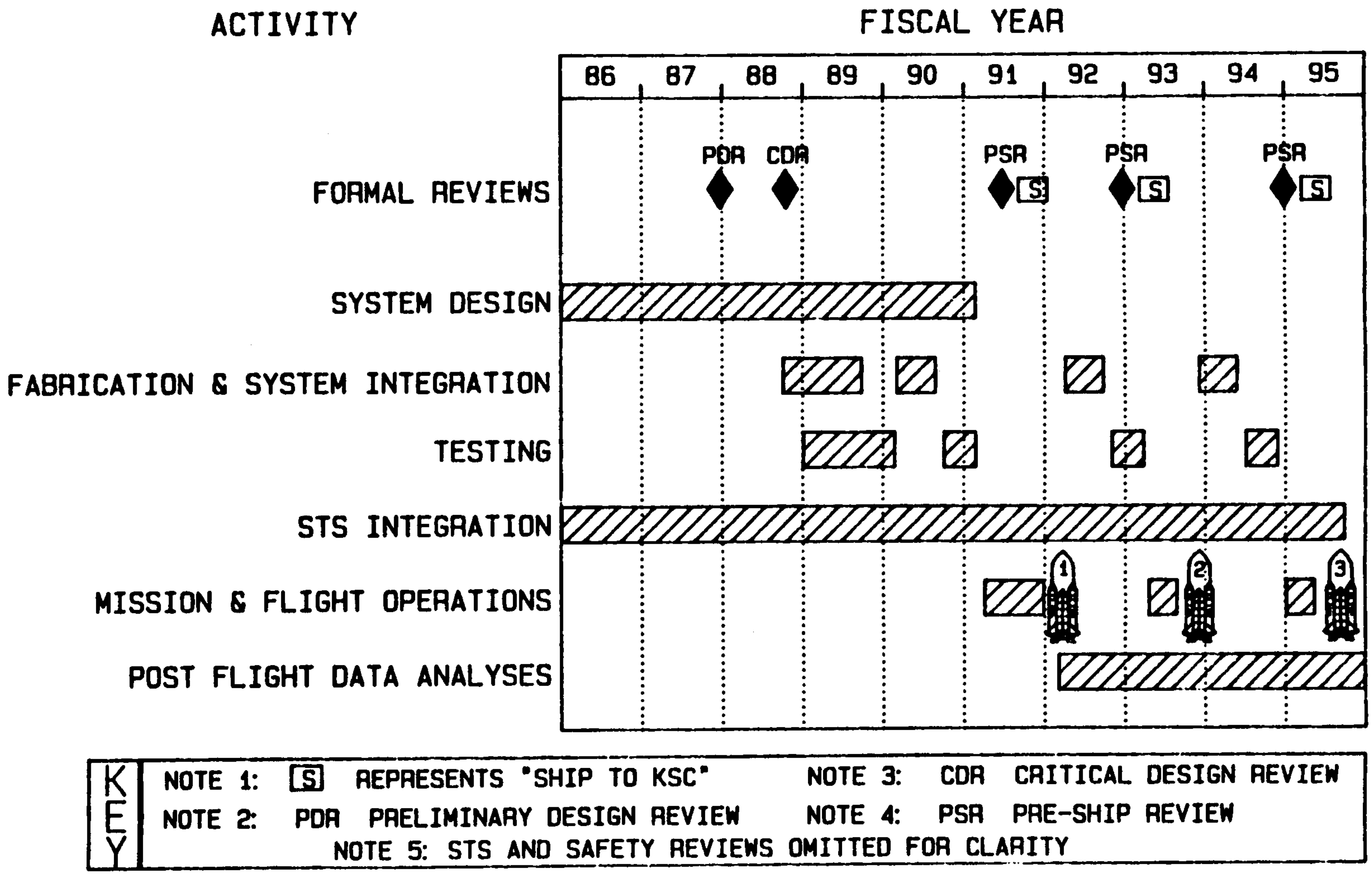


Figure B-1. CFMFE PROGRAM SCHEDULE

FUTURE CRYOGENIC DEVELOPMENT

PROGRAM TITLE: Oxford Stirling Cycle Cooler

CRYOGENIC TECHNOLOGY: Cryogenic Refrigerators

SPONSORING AGENT: NASA - Goddard Space Flight Center

PROGRAM OBJECTIVE: Develop a three-year lifetime Stirling Cycle Cooler for cooling of Infra-Red sensors on the NASA Upper Atmosphere Research Satellite (UARS) to be launched in 1989. The refrigerator must provide one watt of cooling at 80K and use less than 80 watts of electrical input power.

EXPECTED CRYOGENIC DEVELOPMENT: By 1990, a reliable long-life 80K Stirling Cycle Cooler will be operational. A multi-stage machine capable of cooling at lower temperatures is currently under development at Oxford and should be near or at operational status in the 1990 time-frame.

FUTURE CRYOGENIC DEVELOPMENT

PROGRAM TITLE: Long Duration Exposure Facility (LDEF)

CRYOGENIC TECHNOLOGY: Thermal Control Coatings

SPONSORING AGENT: NASA - Langley Research Center

PROGRAM OBJECTIVE: Provide long-term exposure to an on-orbit environment for various experiments that require little or no electrical power and data processing while in space, and which benefit from post-flight laboratory investigation of the retrieved hardware.

EXPECTED CRYOGENIC DEVELOPMENT: LDEF contains numerous samples of thermal control coatings. Upon retrieval, analysis of these samples will provide data on long-term stability of optical properties of these samples. LDEF is scheduled to be flown approximately every 18 months. By the 1990 time-frame, LDEF data from several flights will have provided a large data base on thermal control coating stability. This information will aid in choosing appropriate long lifetime coatings for use on the Long-Term Cryogenic Storage Experiment.

FUTURE CRYOGENIC DEVELOPMENT

PROGRAM TITLE: Lockheed Passive Orbital Disconnect Strut (PODS)

CRYOGENIC TECHNOLOGY: Dual Stage Supports

SPONSORING AGENT: NASA - Ames Research Center

PROGRAM OBJECTIVE: Development of an elastic deformation disconnect strut to lower on-orbit dewar heat leak.

EXPECTED CRYOGENIC DEVELOPMENT: The current PODS-III design has undergone thermal and structural testing. Lockheed considers the PODS-III system ready for flight applications. They are currently developing a PODS-IV version for application on large tankage systems. PODS-III is currently baselined for use on the Space Infrared Telescope Facility (SIRTF). By the 1990 time frame, PODS should be flight qualified and suitable for application in the long-term storage experiment.

FUTURE CRYOGENIC DEVELOPMENT

PROGRAM TITLE: Multi-Layer Insulation Thick Blanket Performance Demonstration

CRYOGENIC TECHNOLOGY: Thick MLI Blankets

SPONSORING AGENT: Air Force Rocket Propulsion Lab (AFRPL)

PROGRAM OBJECTIVE: This program, currently underway at Beech Aircraft, shall characterize and demonstrate the physical and thermal performance of thick MLI blankets (over 2 inches thick and more than 150 layers) on meaningful scale cryogenic fluid storage tanks under simulated launch and space environments. Techniques for fabricating and installing the thick MLI blankets onto large tanks shall be developed and demonstrated.

EXPECTED CRYOGENIC DEVELOPMENT: This program will develop effective attachment and layup techniques for thick MLI blankets. In addition, thermal and structural performance of these systems under simulated launch and space environments will be characterized. This will allow thick MLI blankets to be designed for flight with a high level of confidence in predicted performance.

FUTURE CRYOGENIC DEVELOPMENT

PROGRAM TITLE: Sorption Refrigerator

CRYOGENIC TECHNOLOGY: Active Refrigeration

SPONSORING AGENT: Air Force Wright Aeronautical Laboratories (AFWAL)

PROGRAM OBJECTIVE: Design, build and test a prototype refrigeration unit utilizing gas absorption/absorption compressors that will provide 5 watts of cooling at 7K.

EXPECTED CRYOGENIC DEVELOPMENT: By 1990, a prototype sorption refrigerator will be developed and tested. Development of a flight unit would then be possible based on the prototype design.

FUTURE CRYOGENIC DEVELOPMENT

PROGRAM TITLE: Metal Hydride Test Bed

CRYOGENIC TECHNOLOGY: Metal Hydride H_2 Storage

SPONSORING AGENT: NASA - Marshall Space Flight Center

PROGRAM OBJECTIVE: Perform a thorough investigation of hydride technology, including comparison of hydride vs. cryogenic H_2 storage. Perform materials testing to support design of a metal hydride H_2 storage test bed. Design, build and test a metal hydride storage system to evaluate system performance and optimal component design.

EXPECTED CRYOGENIC DEVELOPMENT: By 1987, a working metal hydride storage system will have been developed and tested. This test bed will allow determination of proper materials and components for use in such systems and characterization of system performance.

FUTURE CRYOGENIC DEVELOPMENT

PROGRAM TITLE: Multi-Stage Magnetic Refrigerator

CRYOGENIC TECHNOLOGY: Cryogenic Refrigerators

SPONSORING AGENT: Strategic Defense Initiative Office - Air Force Space Technology Center, Kirtland AFB, New Mexico

PROGRAM OBJECTIVE: Develop a multi-stage "proof-of-principle" magnetic refrigerator operating in the 4K to 60K temperature range.

EXPECTED CRYOGENIC DEVELOPMENT: This program will develop an operating magnetic refrigerator that can provide cooling at 4K and dump waste heat at 60K. Lifetime and reliability testing, along with coupling this device to a 60K-300K range refrigerator, will be necessary prior to flight applications.

FUTURE CRYOGENIC DEVELOPMENT

PROGRAM TITLE: Compact Cryogenic Feed System Demonstration (CCFSD)

CRYOGENIC TECHNOLOGY: Liquid Acquisition Device, Thermodynamic Vent System, Soft Outer Shell

SPONSORING AGENT: Air Force Rocket Propulsion Laboratory

PROGRAM OBJECTIVE: Perform a preliminary design of a cryogenic upper stage vehicle that utilizes a toroidal LO_2 oxidizer tank. Design, manufacture and test a full scale prototype of the toroidal oxidizer tank.

EXPECTED CRYOGENIC DEVELOPMENT: Beech Aircraft will develop manufacturing techniques required to build a thin-wall aluminum toroidal liquid oxygen tank. A low-g Liquid Acquisition Device (LAD) will be developed and ground-tested with the prototype tank. A coupled thermodynamic vent system, utilizing LH_2 boiloff to cool the tank and LAD will also be developed and tested. A soft outer shell insulation system will be developed and utilized to insulate the prototype tank.

APPENDIX C

AVAILABLE HARDWARE REVIEW

AVAILABLE HARDWARE REVIEW

Hardware: Oxygen Thermal Test Article (OTTA)

Availability: The OTTA is currently in storage at NASA Johnson Space Center and would be available for use.

Description: The OTTA is a high performance 6.45 m^3 (228 ft^3) oxygen dewar built by Beech Aircraft in 1969 for NASA under contract number NASA-10348. See Figure C-1 for further information.

Potential Application: The OTTA could be used as the test dewar in Phases I and III and the supply dewar in Phase II. OTTA has been previously tested utilizing LH_2.

Critical Specifications:

- o Volume 6.45 m^3 (228 ft^3)
- o Spherical Dewar 2.16 m (85 inch) O.D.
- o Design Pressure 1 MPa (150 psia) maximum
- o Two vapor cooled shields
- o 46 Layers Double Silverized Mylar MLI
- o Strap Support System (designed for oxygen)
- o Heat Leak - 1.3 watts (4.45 BTU/Hr) using LH_2
- o Boiloff - 0.056% per day
- o Wet Weight - 2540 kg (5600 lbm) filled with LH_2

Advantages

- o Existing hardware will reduce costs and development time
- o Dewar is approximately the volume required for scalability and has been tested using LH_2

Disadvantages

- o Pressure vessel has experienced corrosion during storage

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OXYGEN THERMAL TEST ARTICLE - OTTA

The Boulder Division, under direct contract to NASA, designed and fabricated a prototype spherical tank for space storage of cryogenic fluids, either liquid oxygen, hydrogen, nitrogen, helium, or methane. Originally the Oxygen Thermal Test Article (OTTA) and Hydrogen Thermal Test Article (HTTA) were prototype supply tank designs for a Space Shuttle cryogenic fueled Orbital Maneuvering System (OMS).

The overall tank diameter is seven feet (2.16 meters). The pressure vessel is suspended from the outer shell by "E" glass straps. Three separate straps encircle the pressure vessel. Each strap is attached on opposite sides to the girth ring and is supported off the pressure vessel by insulating pads. The annulus between the pressure vessel and outer shell contains both a boiler shield and a vapor-cooled shield.

The test results of tank performance far surpassed proposed results. Tests completed with liquid hydrogen showed a heat leak of 4.45 Btu/Hour (1.30 watts) or a daily loss rate of 0.056 percent. Helium tests showed a heat leak of 1.22 Btu/hour (0.36 watts) or 0.21 percent daily loss rate.

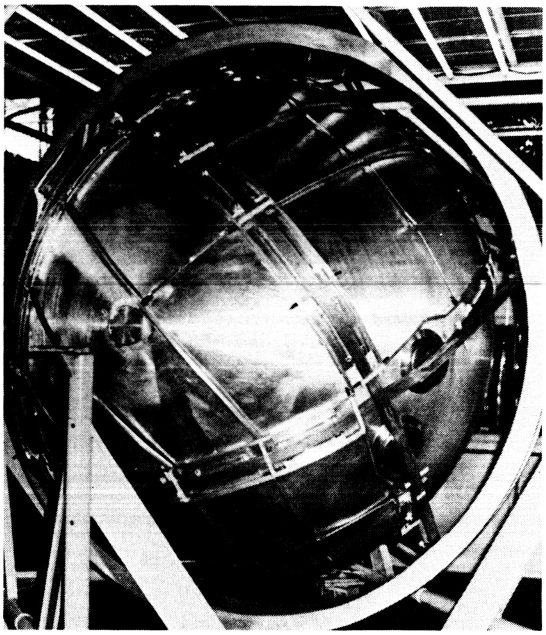


Figure C-1.

OTTA DESCRIPTION.

FLUID:	LIQUID OXYGEN
CONFIGURATION:	DOUBLE-WALL SPHERICAL VESSEL
OVERALL DIMENSIONS:	84.94 IN (215.75 CM) DIAMETER
CAPACITY:	228.34 FT^3 (6.47 M^3)
PRESSURE:	49.4 PSIA (34.06 N/CM^2)
MATERIAL:	
PRESSURE VESSEL:	2219 ALUMINUM
OUTER SHELL:	6061 ALUMINUM
ENVIRONMENT:	SPACE (GROUND TEST)
LOADS:	7 G VERTICAL, 3 G HORIZONTAL
ANNULUS VACUUM:	10^{-6} TORR
INSULATION SYSTEM:	MULTILAYER, VAPOR-COOLED SHIELD AND VACUUM
WEIGHT (DRY):	4595.0 LB (2083.9 KG)
THERMAL PERFORMANCE:	HEAT LEAK: 8.4 BTU/HR (2.4 WATTS) PREDICTED MAXIMUM
PIPING SIZES:	
FEED/FILL:	.875 x .020 W (22.22 x .51 mm)
VENT:	.875 x .020 W (22.22 x .51 mm)
VCS:	.187 x .028 W (4.75 x .71 mm)

Recommendation (including required modifications):

OTTA would require the following modifications for use in the experiment:

- Extensive cleaning or rebuild of pressure vessel.
- Addition of thermodynamic vent system heat exchangers and Joule - Thomson valve.
- Addition of more multi-layer insulation.
- Replacement of suspension system to dual stage supports.
- Addition of fluid acquisition device.

OTTA could possibly be used in the LTCFSE experiment. Further investigation into the condition of the tank is required to make such a decision. At a minimum, the OTTA design could be utilized and modified as required.

AVAILABLE HARDWARE REVIEW

Hardware: Hydrogen Thermal Test Article (HTTA)

Availability: The HTTA is currently in storage at Rocketdyne and would be available for use.

Description: The HTTA is a high performance 227 m3 (800 ft3) hydrogen dewar built by Beech Aircraft in 1972 for NASA under contract number NASA-12105. See Figure C-2 for further information.

Potential Application: The HTTA could be used as the test dewar in Phases I and III and the supply dewar in Phase II.

Critical Specifications:

- o Volume - 22.7 m3 (800 ft3)
- o Cylindrical Dewar, 6.64 m (21.8 ft) long x 2.8 m (9.2 ft) diameter
- o Design Pressure 172-345 kPa (25-50 psia)
- o Two Vapor Cooled Shields
- o 54 layers double silverized Mylar MLI
- o Strap Support System
- o Heat Leak - 1.9 watts (6.4 BTU/hr)
- o Boiloff - 0.022% per day
- o Wet Weight - 3730 kg (8,220 lbm) filled with LH_2

Advantages

- o Existing hardware will reduce costs and development time.
- o Dewar is a large, high performance H2 tank in good condition.

Disadvantages

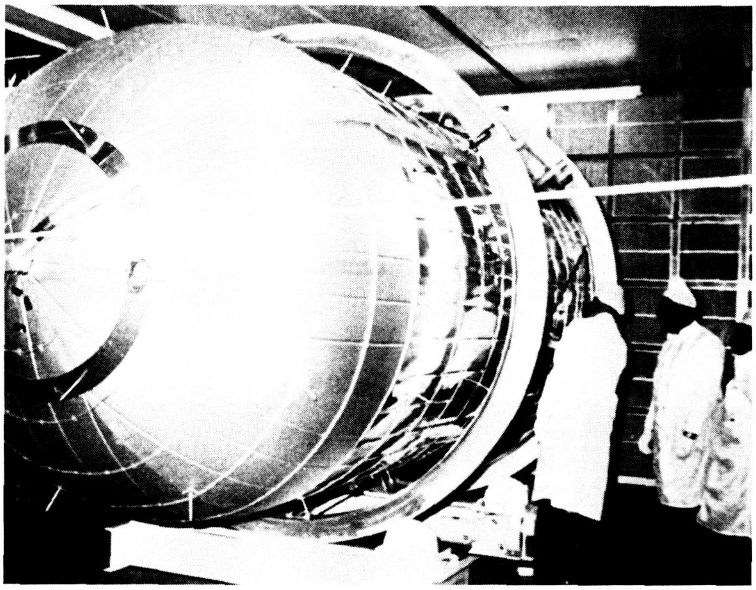
- o Outer Shell is not flight weight
- o Current annular vacuum acquisition problem would most likely require buildup of new pressure vessel.
- o Dewar is larger than needed for test.

HYDROGEN THERMAL TEST ARTICLE - HTTA

The Boulder Division, under direct contract to NASA-JSC, designed and built the Hydrogen Thermal Test Article (HTTA) - a prototype tank for long duration cryogenic space storage.

The overall tank dimensions are 21.8 feet (6.6 meters) in length by 9.7 feet (2.96 meters) in diameter. The 800 cubic foot (22.7 cubic meters) volume holds 3,500 pounds (1587 kg) of liquid hydrogen at subcritical pressure for 180 days with minimum fluid loss. The evacuated annulus contains multi-layer insulation. An active vapor-cooled shield circulates boil-off hydrogen gas to reduce the heat leak from 17.1 Btu/hour (5.0 watts) and no vapor cooling to 6.40 Btu/hour (1.9 watts) with vapor cooling. Fiberglass straps suspend the inner pressure vessel from the outer shell girth ring. The straps support the weight of the hydrogen as well as prevent excessive heat leak.

The HTTA has also been used by Rockwell International for a helium pump test. Due to its large volume and large outflow line, the HTTA supplied a high flow rate of helium to the helium pump.



FLUID:	LIQUID HYDROGEN
CONFIGURATION:	DOUBLE-WALL CYLINDRICAL VESSEL
OVERALL DIMENSIONS:	9.7 FT (2.96 M) DIAMETER X 21.8 FEET (6.6 METERS) IN LENGTH
CAPACITY:	800 FT^3 (22.7 M^3)
PRESSURE:	50 PSIA (34.5 N/CM^2)
MATERIAL:	
PRESSURE VESSEL:	2219 ALUMINUM
OUTER SHELL:	2219 ALUMINUM
ENVIRONMENT:	SPACE (GROUND TEST)
LOADS:	3.3 G VERTICAL, 1 G HORIZONTAL
ANNULUS VACUUM:	10^{-5} TORR
INSULATION SYSTEM:	MULTILAYER, VAPOR-COOLED SHIELD AND VACUUM
WEIGHT (DRY):	4700 LB (2131.5 KG)
THERMAL PERFORMANCE:	HEAT FLUX: 17.1 BTU/HR (5.0 WATTS) MAXIMUM PREDICTED
PIPING SIZES:	
FEED/FILL:	2.5 x .028 W (63.5 x .71 mm)
VENT:	2.5 x .028 W (63.5 x .71 mm)
VCS:	.1875 x .028 W (4.75 x .71 mm)

Figure C-2.

HTTA DESCRIPTION.

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Recommendation (including required modifications):

HTTA would require fairly extensive modifications including:

- o Addition of thermodynamic vent system heat exchangers and Joule-Thomson Valve.
- o Rebuild pressure vessel.
- o Addition of more multilayer insulation.
- o Rebuild outer shell to flight weight and configuration.
- o Replacement of suspension system.
- o Addition of fluid acquisition device.

Due to these considerations, and of the fact that the dewar is larger than required, HTTA is not a good candidate for use in the experiment. Extensively modifying the tank, by removing some or all of the cylindrical section of the tank would make HTTA a more desirable candidate.

AVAILABLE HARDWARE REVIEW

Hardware: Cryogenic Fluid Management Flight Experiment (CFMFE) receiver tank.

Availability: The CFMFE experiment is currently in the design phase. The hardware is to be flown by the early 1990s and will be available in time for use in the Long-Term Storage Experiment.

Description: The CFMFE receiver tank is a 0.175 scale OTV tank designed to test no vent filling from a larger supply tank.

Potential Application: Use as the receiver tank for the Phase II fluid transfer portion of the LTCFSE experiment.

Critical Specifications:

- Volume - 0.38 m3 (13.4 ft3)
- Design Pressure - 414 kPa (60 psia)
- Tangential and radial spray nozzles mounted in tank for pre-chill tank cooldown and tank fill.
- Thermodynamic vent system
- 60 Layers Double Aluminized Mylar MLI
- Externally mounted heaters available for thermal conditioning
- Cylindrical tank - 1.04 m (40.9 in.) long x .51 m (20.0 in.) diameter.

Advantages

- o Tank contains necessary instrumentation needed for test.
- o Tank will be flight qualified prior to LTCFSE test.
- o Tank is scalable to OTV dimensions.
- o Tank contains spray nozzles needed for tank cooldown and fill.
- o Use of existing hardware will reduce cost and development time.

Disadvantages

- o Time frame of CFMFE flights could cause an availability problem.
- o Experiment data would duplicate CFMFE results.

Recommendation (including required modifications):

Minimal modification would be required for use on the LTCFSE experiment. However, since use of this tank would duplicate CFMFE results, it is not recommended for use.

AVAILABLE HARDWARE REVIEW

Hardware: Fuel Cell Servicing System (FCSS)

Availability: The FCSS is in place at Cape Canaveral and Vandenburg AFB launch facilities and is available for use.

Description: FCSS is a liquid hydrogen and oxygen transfer and pressurization system used to fill and pressurize the Shuttle Power Reactant Supply Assembly (PRSA) tanks.

Potential Application: The FCSS can be used to load the experiment dewar prior to launch. It is currently baselined for loading the CFMFE.

Critical Specifications: See Figure C-3.

Advantages

- System is currently being used for Shuttle flights, and has proven performance.
- System requires no modification for use, minor modification to the Shuttle required.
- High purity LH2 is utilized in system.

Disadvantages

- Additional LH2 tank truck must be connected to system to provide adequate quantities of LH2 to load the LTCFSE experiment.

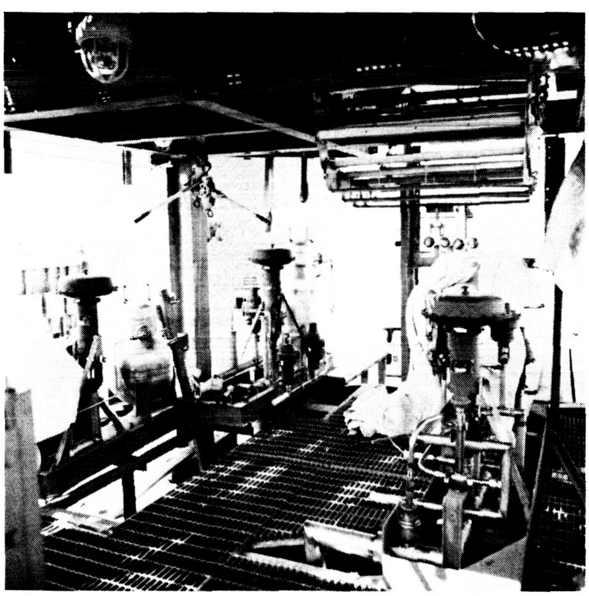
Recommendation (including required modifications):

The FCSS could be used to fill the experiment dewar by adding a tee line and valve to one of the Shuttle PRSA fill lines. It is recommended that the FCSS be utilized to perform experiment fill prior to launch.

FUEL CELL SERVICING SYSTEM - FCSS

The Boulder Division, under contract to NASA-KSC, completed the design, development, fabrication, certification, and delivery of the Space Shuttle Vehicle Fuel Cell Servicing System (FCSS). The FCSS 2000-gallon (7.57 cubic meter) liquid hydrogen dewar, 800-gallon (3.03 cubic meter) liquid oxygen dewar and low-pressure valve and relief complex is located on the 155-foot level of the Fixed Service Structure (FSS) at the Kennedy Space Center launch site. The required flowrate of hydrogen is 40 gallons per minute (151 liters per minute) at a minimun density of 4.37 pounds per cubic foot (.070 grams per cubic centimeter). The required flowrate of oxygen is 20 gallons per minute (76 liters per minute) at a minimum density of 70.65 pounds per cubic foot (1.13 grams per cubic centimeter). The vacuum-jacketed fill and vent lines extend to the Rotating Service Structure (RSS). The high pressure valve and accumulator complex on the RSS is capable of hydraulically pressurizing the oxygen tanks in the Space Shuttle Vehicle to 1,050 psi (724 N/cm^2) and the hydrogen tanks to 350 psi (241 N/cm^2).

The FCSS was built from commercially available parts procured from over 40 different vendors. The system is designed for a 10-year service life. The FCSS at Launch Complex LC39A has serviced all the early Space Shuttle launches from Kennedy Space Center. Two additional systems have been produced for Launch Complex LC39B and for the launch facility at Vandenberg Air Force Base, California.



FUEL CELL SERVICING SYSTEM

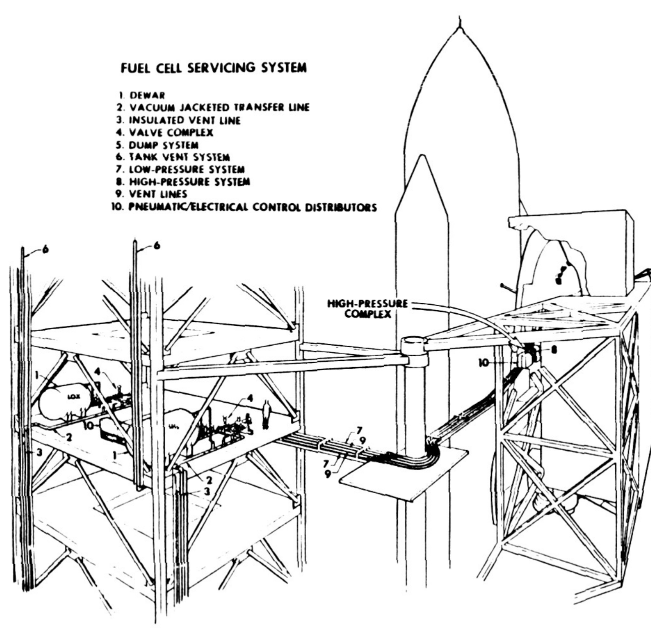


Figure C-3. FCSS DESCRIPTION.

AVAILABLE HARDWARE REVIEW

Hardware: Cryogenic Fluid Management Flight Experiment (CFMFE) Supply Tank.

Availability: The CFMFE is currently in the design phase. The hardware is to be flown by the early 1990s and may be available in time for use in the Long-Term Storage Experiment.

Description: The CFMFE supply tank is a 0.6 m3 (21.2 ft3) spherical hydrogen dewar containing a liquid acquisition device for low-g expulsion of liquid.

Potential Application: The CFMFE supply tank could be used as the experiment dewar in Phase I and III and as the supply dewar in Phase II.

Critical Specifications:

- o Volume 0.6 m3 (21.2 ft3)
- o Design Pressure 414 kPa (60 psia)
- o Thermodynamic Vent System with vapor cooled shield
- o Liquid Acquisition Device (LAD) mounted inside PV
- o MLI - 135 Layers Double aluminimized Mylar with double Dacron B4A spacers
- o Trunnion Pressure Vessel Support

Advantages

- o Use of existing hardware will reduce cost and development time
- o Tank configuration (ie: LAD, TVS, thick MLI) is well suited for application
- o Tank will be flight-qualified prior to LTCFSE experiment

Disadvantages

- o Tank is much smaller than desired for experiment
- o Time frame of CFMFE flights could cause an availability problem
- o Tank would require extensive modification for use with dual stage supports

Recommendation (including required modifications):

Required Modifications:

- Addition of para-to-ortho H_2 converter
- Addition of dual stage supports

The CFMFE supply tank is not an attractive candidate for the Long-Term Cryogenic Storage Experiment, primarily due to the small size of the tank.

AVAILABLE HARDWARE REVIEW

Hardware: Earth Limb Measurement Satellite (ELMS) dewar.

Availability: The ELMS tank is currently in storage at Beech Aircraft in Boulder, Colorado and is available for use.

Description: The ELMS tank is a 1.27 m3 (45 ft3), supercritical helium dewar. It is a cylindrical tank with hemispherical heads. For more information, see Figure C-4.

Potential Application: The ELMS tank could be used as the experiment dewar or as the receiver tank in Phase II.

Critical Specifications:

- o Volume 1.27 m^3 (45 ft^3)
- o One Vapor Cooled Shield
- o Fiberglass Strut Support System
- o Design Pressure 607 kPa (88 psia)
- o Dry Weight 159 kg (350 lb)
- o Heat Leak 1.5 watts (5.0 BTU/hr) using LH_2
- o Nine layers double aluminized/nylon net MLI

Advantages

- o Existing hardware will reduce costs and development time.
- o Tank is flight weight.
- o Tank would scale to OTV volumes better than CFMFE receiver tank, pressure vessel mass to volume ratio, m/v, is 33.6 kg/m^3 (2.1 lbm/ft^3)

Disadvantages

- o Tank is smaller than required for use as main experiment dewar.

HELIUM TEST TANK - ELMS

The Boulder Division, under contract to Grumman Aerospace, designed and produced a cylindrical, double-walled tank to contain supercritical helium in a spacecraft which was originally the Earth Limb Measurement Satellite (ELMS). The helium was to be used to maintain an infrared sensor at a temperature of 18° R (10° K).

The tank volume is 45 cubic feet (1.27 cubic meters) and stores up to 410 pounds (186 kg) of helium. The pressure vessel is supported by 10 high strength/low conductivity tubular fiberglass struts. The struts extend from the outer shell girth ring to bosses on the pressure vessel. The insulation is perforated double aluminized mylar with nylon net spacers. A vacuum ionization pump is available to monitor the annulus vacuum.

The thermal performance of the tank is varied by controlling the amount of helium gas circulating through the vapor-cooled shield (VCS). Development of a microprocessor-based controller was funded by Beech IR&D to provide control of the VCS flow.

FLUID:	SUPERCRITICAL HELIUM
CONFIGURATION:	DOUBLE-WALL CYLINDRICAL VESSEL WITH HEMISPHERICAL HEADS, FIBERGLASS STRUTS
OVERALL DIMENSIONS:	84.4 IN (214.38 CM) LENGTH x 48.2 IN (122.43 CM) DIAMETER
CAPACITY:	45 FT^3 (1.27 M^3)
PRESSURE:	88 PSIA (60.7 N/CM^2)
MATERIAL	
PRESSURE VESSEL:	2219 ALUMINUM
OUTER SHELL:	6061 T6 ALUMINUM
ENVIRONMENT:	SPACE
LOADS:	+11.3, -1.3 G AXIAL, $\pm$2.5 G LATERAL
ANNULUS VACUUM:	5 x 10^{-5} TORR
INSULATION SYSTEM:	MULTILAYER, VAPOR-COOLED SHIELD, VACUUM
WEIGHT (DRY):	350 LB (158.7 KG)
THERMAL PERFORMANCE:	HEAT LEAK VARIABLE 39.7 BTU/HR (11.5 WATTS) MAXIMUM PREDICTED
PIPING SIZES:	
FEED:	.25 IN (6.35 mm)
FILL:	.5 IN (12.70 mm)
VENT:	.625 x .020 W (15.88 x .51 mm)
VCS:	.1875 x .028 W (4.76 x .71 mm)

Figure C-4. ELMS DESCRIPTION.

Recommendation (including required modifications):

Required Modifications:

- o Addition of liquid acquisition device
- o Addition of thermodynamic vent system
- o Addition of more MLI
- o Addition of liquid spray nozzles (if used as receiver tank)
- o Addition of more instrumentation
- o Remove vapor cooled shield

The ELMS tank is an attractive candidate for use as the Phase II receiver tank. Due to its relatively small size, the ELMS tank is not suitable for use as the main experiment dewar.

AVAILABLE HARDWARE REVIEW

Hardware: Centaur GSE Loading System

Availability: The Centaur Loading System is currently available and operational on Pad 39A at KSC.

Description: The Centaur Loading System supplies LH_2 and LO_2 to the Centaur Upper Stage prior to STS launch. This system utilizes the STS external tank cryogenic supply dewars as the source of LH_2 and LO_2, and provides fill and drain operations through the shuttle T-0 umbilical panel.

Potential Application: The Centaur Loading System could be utilized to fill the LTCFSE supply dewar with LH_2 prior to launch.

Critical Specifications: Not available.

Advantages	Disadvantages
o System has adequate capacity to fill supply tank.	o The Centaur GSE uses a lower grade LH_2 than does the FCSS. This could cause TVS Joule-Thomson valve contamination.
	o Large number of support personnel required for operation.
	o Cancellation of Centaur program makes availability questionable.

Recommendation (including required modifications):

Required Modifications: None

The Centaur GSE is not recommended for use in loading the LTCFSE experiment due to the disadvantages listed above.

AVAILABLE HARDWARE REVIEW

Hardware: Centaur Orbiter Mod Kit

Availability: Orbiter OV-104 (Atlantis) has been modified to use the Mod Kit as required. Availability may be questionable due to cancellation of the Shuttle/Centaur program.

Description: The Centaur Orbiter Mod Kit contains all fluid interfaces required for LH_2/LO_2 fill, drain and dump and LH_2/LO_2 ground, ascent and on-orbit venting. Fluid interfaces are located in the aft end of the payload bay and are designed to interface with the Centaur Integrated Support System (CISS).

Potential Application: The Orbiter Mod Kit could be utilized to provide LH_2 drain, dump and vent capabilities to the LTCFSE supply tank.

Critical Specifications: LH_2 system only:

- H_2 Fill and Drain Line - 4.9 cm (1.93 in) ID
- H_2 Ground Vent Line - 4.9 cm (1.93 in) ID
- H_2 Ascent and Abort Vent Line - 4.9 cm (1.93 in) ID
- H_2 Dump and On-Orbit Vent Line - 13.7 cm (5.4 in) ID

These fluid lines are depicted in Figure C-5.

Advantages	Disadvantages
o Use would eliminate further high cost mods to Orbiter that would produce further scarring.	o None.

Recommendation (including required modifications):

Interface lines with quick-disconnects must be constructed between the LTCFSE experiment and Mod Kit Lines. The Centaur Orbiter Mod Kit is recommended for use.

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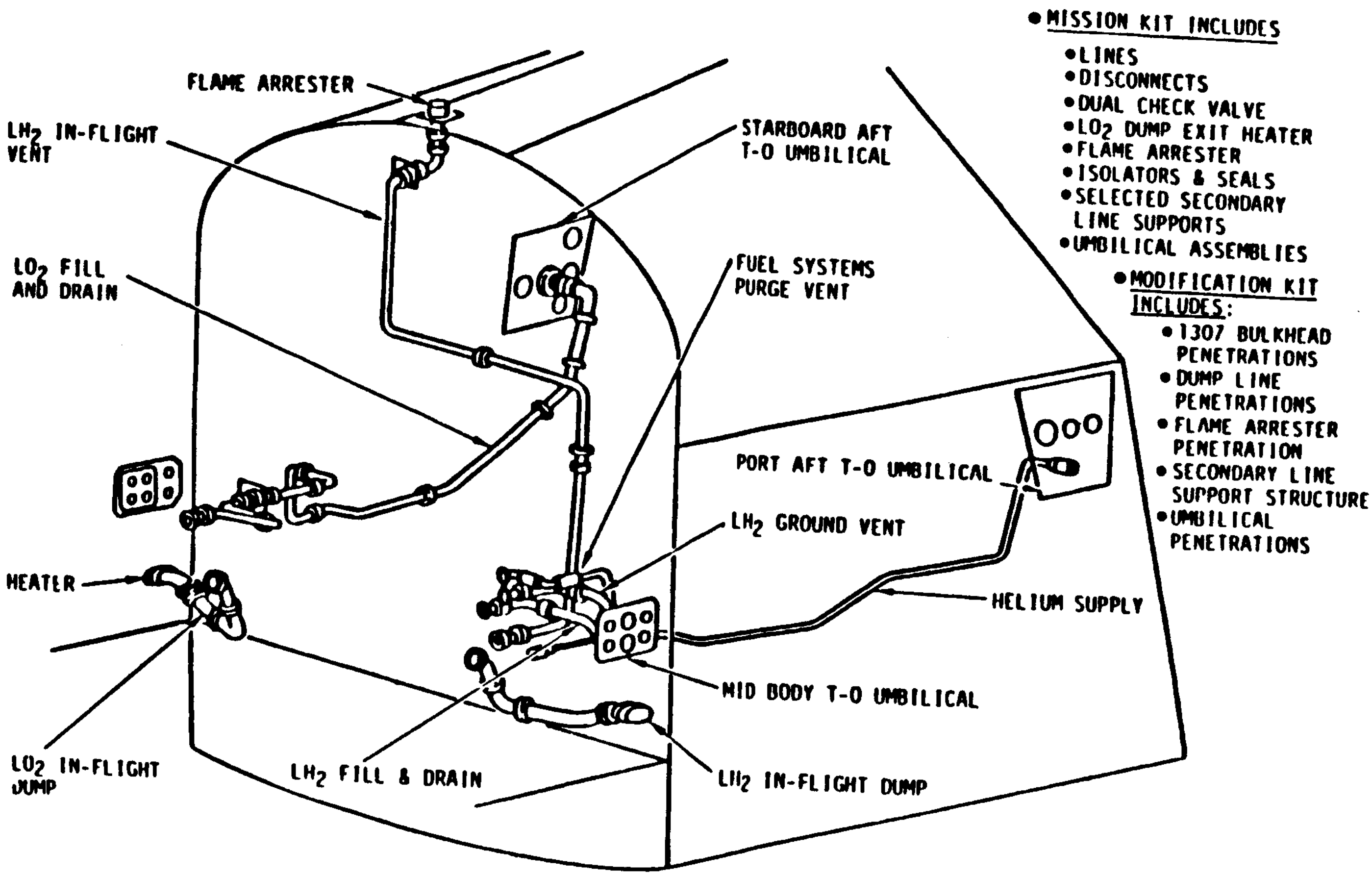

Figure C-5. CENTAUR ORBITER MOD KIT FLUID LINES.

AVAILABLE HARDWARE REVIEW

Hardware: Power Reactant Supply Assembly (PRSA) Hydrogen Tank.

Availability: The PRSA tanks are currently being flown on the Shuttle. They are currently not available for use.

Description: The PRSA hydrogen tank is a .615 m^3 (21.7 ft^3) flight qualified hydrogen dewar. See Figure C-6 for more details.

Potential Application: Use as a receiver tank in Phase II fluid transfer experiments.

Critical Specifications:

- o Volume - .615 m^3 (21.7 ft^3)
- o Design Pressure 1.97 MPa (285 psia)
- o Spherical Dewar - 1.2 m (47.24 in.) O.D.
- o One Vapor Cooled Shield
- o Strap Support System
- o 14 Layers double silverized mylar MLI
- o Heat Leak - 2.6 watts (8.8 BTU/hr)
- o Wet Weight - 146 kg (322 lbm)

Advantages

- o Use of available design will reduce experiment cost and development time.
- o Tank has been Shuttle flight qualified.

Disadvantages

- o Large amount of rebuild needed to reconfigure as a receiver tank, particularly internal to the pressure vessel.
- o Pressure vessel mass to volume ratio (m/V) is higher than desired.

Recommendation (including required modifications):

The following modifications would be required to reconfigure as a suitable receiver tank:

- o Addition of a thermodynamic vent system, external heat exchanger and Joule-Thomson valve.
- o Addition of more MLI
- o Addition of spray nozzle system internal to the pressure vessel.
- o Addition of more instrumentation.
- o Addition of Liquid Acquisition Device

Since the hardware is not available, and PV m/v ratio is higher than desired, utilization of the PRSA H_2 tank is not recommended.

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POWER REACTANT STORAGE ASSEMBLY - PRSA

The Boulder Division supplies the cryogenic Power Reactant Storage Assembly (PRSA) tanks for the NASA Space Shuttle Orbiter as subcontractor to Rockwell International Space Division. Eight PRSA tanks, four supercritical hydrogen and four supercritical oxygen, are furnished for each Orbiter. Electrical power is developed in the fuel cells from the reaction of the hydrogen and oxygen, with potable water as a by-product. The oxygen is also used to maintain the Orbiter cabin breathing requirements.

Components within the pressure vessel are quantity probe, bulk fluid and heater(s) temperature sensor, and one (hydrogen) or two (oxygen) electrical heaters to maintain pressure during expulsion of the supercritical fluid. Within the evacuated annulus are layers of kapton silverized on both sides alternating with layers of nylon net. Fiberglass straps suspend the inner pressure vessel from the girth ring. The straps extend from the girth ring to bosses machined in the pressure vessel. The hydrogen tank is cooled by a vapor-cooled shield to achieve the required thermal performance. The girth ring contains a vacuum pinch-off tube, a quantity gaging signal conditioner, three mounting trunnions, an ion pump and pump power supply, a vacuum annulus rupture disc, and upper and lower outer shells.

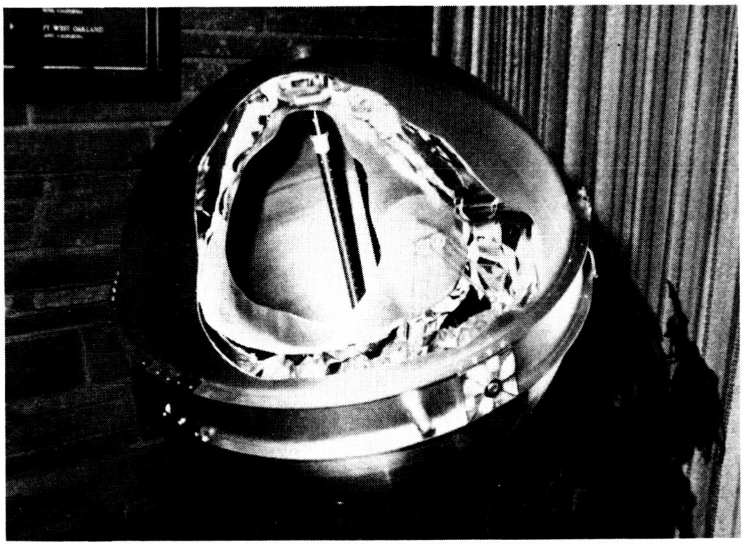


Figure C-6.

PRSA DESCRIPTION.

	PRSA H_2	PRSA O_2
FLUID:	SUPERCRITICAL HYDROGEN	SUPERCRITICAL OXYGEN
CONFIGURATION:	DOUBLE-WALL SPHERICAL VESSEL SUPPORTED BY 12 FIBERGLASS STRAPS	DOUBLE-WALL SPHERICAL VESSEL SUPPORTED BY 12 FIBERGLASS STRAPS
OVERALL DIMENSIONS:	54 IN (137 CM) MAXIMUM ENVELOPE DIAMETER	48 IN (122 CM) MAXIMUM ENVELOPE DIAMETER
CAPACITY:	21.395 FT^3 (.61 M^3)	11.27 FT^3 (.32 M^3)
PRESSURE:	250 PSIA (172 N/$CM^{2)}$	900 PSIA (620 N/CM^2)
MATERIAL:		
PRESSURE VESSEL:	2219 ALUMINUM	INCONEL 718
OUTER SHELL:	2219 ALUMINUM	ALUMINUM 2219
ENVIRONMENT:	SPACE (POWER REACTANT)	SPACE (POWER REACTANT)
LOADS:	$\pm$5G, ACCELERATION AND RANDOM VIBRATION	$\pm$5G ACCELERATION AND RANDOM VIBRATION
ANNULUS VACUUM:	5 x 10^{-4} TORR TO 10^{-7} TORR	5 x 10^{-4} TORR TO 10^{-7} TORR
INSULATION SYSTEM:	MULTILAYER, VAPOR-COOLED SHIELD AND VACUUM	MULTILAYER, VACUUM ANNULUS
WEIGHT (DRY):	227 LB (103 KG)	215 LB (97.5 KG)
THERMAL PERFORMANCE:	HEAT LEAK = 8.5 BTU/HR (2.5 WATTS) ACTUAL	HEAT LEAK = 21 BTU/HR (6.2 WATTS) ACTUAL
PIPING SIZES:		
FEED:	.250 x .020 W (6.35 x .51 mm)	.250 x .020 W (6.35 x .51 mm)
FILL:	.500 x .020 W (12.70 x .51 mm)	.375 x .020 W (9.53 x .51 mm)
VENT:	.500 x .020 W (12.70 x .51 mm)	.500 x .028 W (12.70 x .71 mm)

APPENDIX D

TAG FORMS

TECHNOLOGY DEVELOPMENT EXPERIMENT DESCRIPTION INSTRUCTIONS

The Technology Development Advocacy Group (TDAG) of the Space Station Task Force is conducting an activity to better define potential technology missions for Space Station.

Technology development experiments are defined as research projects performed on Space Station which will provide the technological basis for expanding and improving Space Station capability or for developing commercial products which utilize in-space fabrication or processing. Proposed experiments should have the following characteristics:

(1) Space Station is essential for the accomplishment of experimental objectives. Unique requirements may include long durations in space, availability of power, or availability of large spatial areas.

(2) The technology is appropriate for the 1991 to 2000 time frame. Experiments should be aimed at projected future needs and capability. Experiments may be performed within a laboratory module, as an attachment, or on a co-orbiting platform.

The attached experiment description questionnaire is designed to assist you in providing the TDAG with a preliminary conceptual design of your proposed experiment. This information will be used in planning activities for Space Station and as a basis for the incorporation of user requirements in the Phase B preliminary design activity. Please answer each question as completely as possible using additional sheets if the space provided is inadequate and return a typewritten copy to your TDAG representative. Questions marked with an asterisk will serve as background information for review by OAST management. More detailed conceptual designs and precursor program plans may be requested for certain experiments in the future as planning activities progress.

TECHNOLOGY EXPERIMENT DESCRIPTION

PROPOSER: NAME: Mr. Roger Scarlotti

ADDRESS: Beech Aircraft
P. O. Box 9631
Boulder, CO 80301

PHONE: (303) 443-1650

TDAG CONTACT: NAME:

ADDRESS:

PHONE:

TDM CATEGORY: TDMX2311

MRWG NO.:

GENERAL

*1. Briefly describe the mission objective.

The Long-Term Cryogenic Fluid Storage (LTCFSE) experiment will demonstrate long-term storage and transfer of cryogenic fluids in an on-orbit environment. Various technologies utilized in long-term storage and transfer of cryogens will be tested and evaluated. These technologies range from basic passive technologies, such as thick multi-layer insulation blankets, to active refrigeration systems.

*2. What are the potential benefits?

This experiment will perform on-orbit testing and evaluation of technologies required for long-term storage and transfer of cryogenic fluids. These technologies are necessary for operation and resupply of orbital transfer vehicles and other future on-orbit cryogenic applications.

*3. Why is Space Station necessary for accomplishment of the objectives? Specifically, what Space Station characteristics are essential or highly beneficial?

A long-term low-g environment is necessary to evaluate performance of many of the technologies to be investigated. In addition, it is highly desirable to test the long-term effect the harsh orbital environment (e.g. thermal, micrometeroid, atomic oxygen) has on experiment performance. Space Station electrical power, data acquisition, and cooling systems are required for experiment support.

*4. How is the experiment related to ongoing or planned programs?

The experiment will expand on and utilize information obtained in the Cryogenic Fluid Management Flight Experiment (CFMFE) to be flown on Shuttle in the early 1990s. The thrust of this experiment will be on long orbital duration testing as opposed to the shorter term ($\leq$ 1 week) CFMFE. Information gained in this experiment will be utilized in design of future orbital cryogenic applications, such as space-based Orbital Transfer Vehicles and its associated in-space servicing facility.

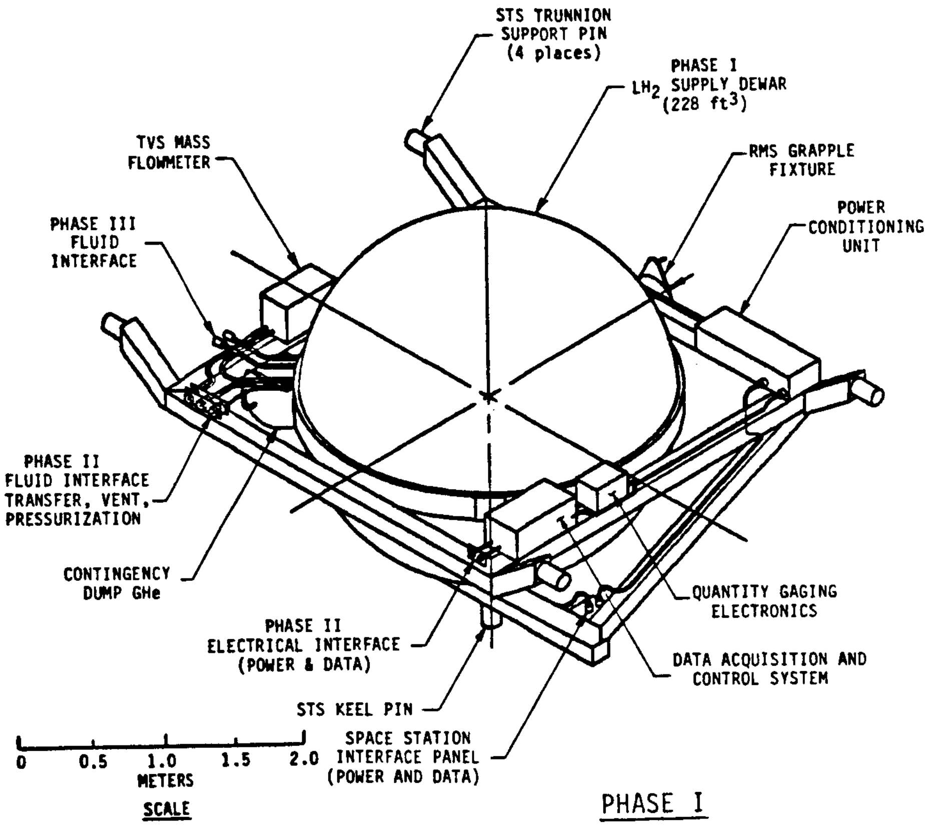
*5. Describe the experiment. What we propose to do, how, and when? (Include suggested flight dates and time phasing rationale).

The experiment consists of three phases, each to take place on Space Station. Phase I will test and evaluate basic passive technologies utilized in long-term cryogenic storage. Currently scheduled for a 1993 deployment, Phase I will consist of a 6.5 m^3 (228 ft^3) liquid hydrogen tank and its associated support structure, data acquisition and control system, and interface hardware. Once mounted to Space Station, the dewar will be allowed to reach thermal equilibrium. Thermal performance data during the 2-year test will be recorded and downlinked to earth.

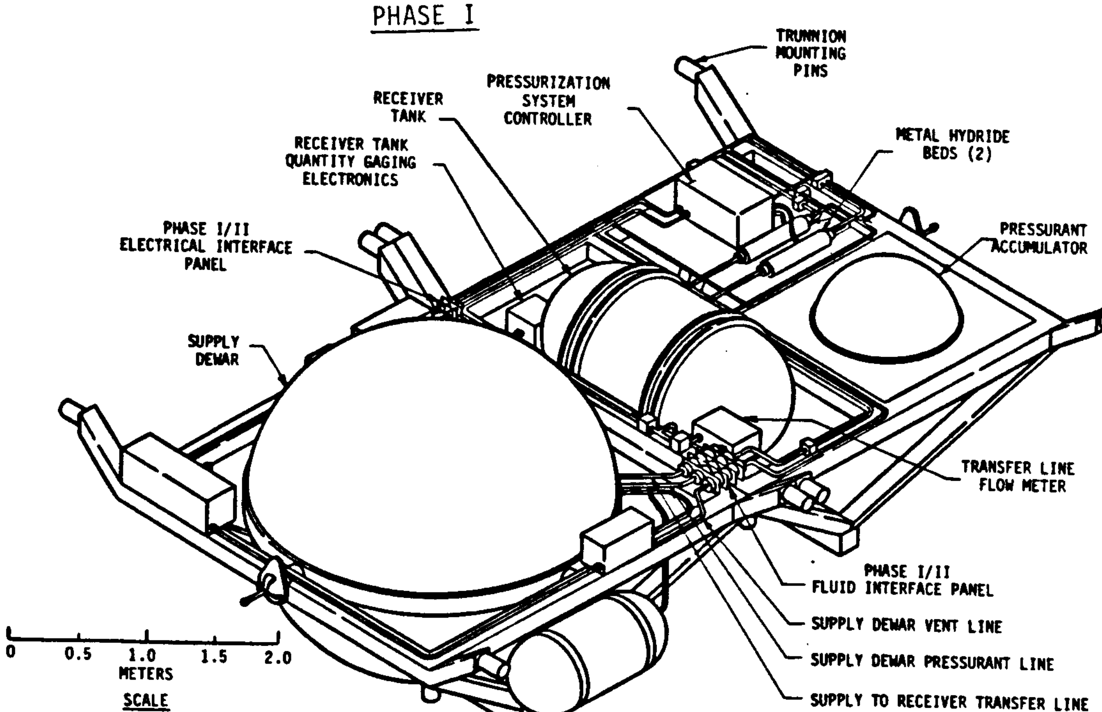
Phase II of the experiment will test and evaluate fluid transfer technologies. The hardware will be reconfigured on-orbit by adding a 1.3 m^3 (45 ft^3) receiver tank, transfer lines, and pressurization system. The Phase II hardware will be deployed on Space Station in 1995. Approximately 10 fluid transfer experiments will occur over a 1-year time frame. The equipment will remain on Space Station for Phase III testing.

Phase III of the experiment will test and evaluate active refrigeration technology. The hardware will be reconfigured on-orbit by adding a pallet containing the refrigerator and required support equipment to the Phase II hardware. The refrigerator system will provide refrigeration to the Phase I liquid hydrogen tank and reduce or eliminate tank boiloff. This system will be allowed to run steady state for a period of one year.

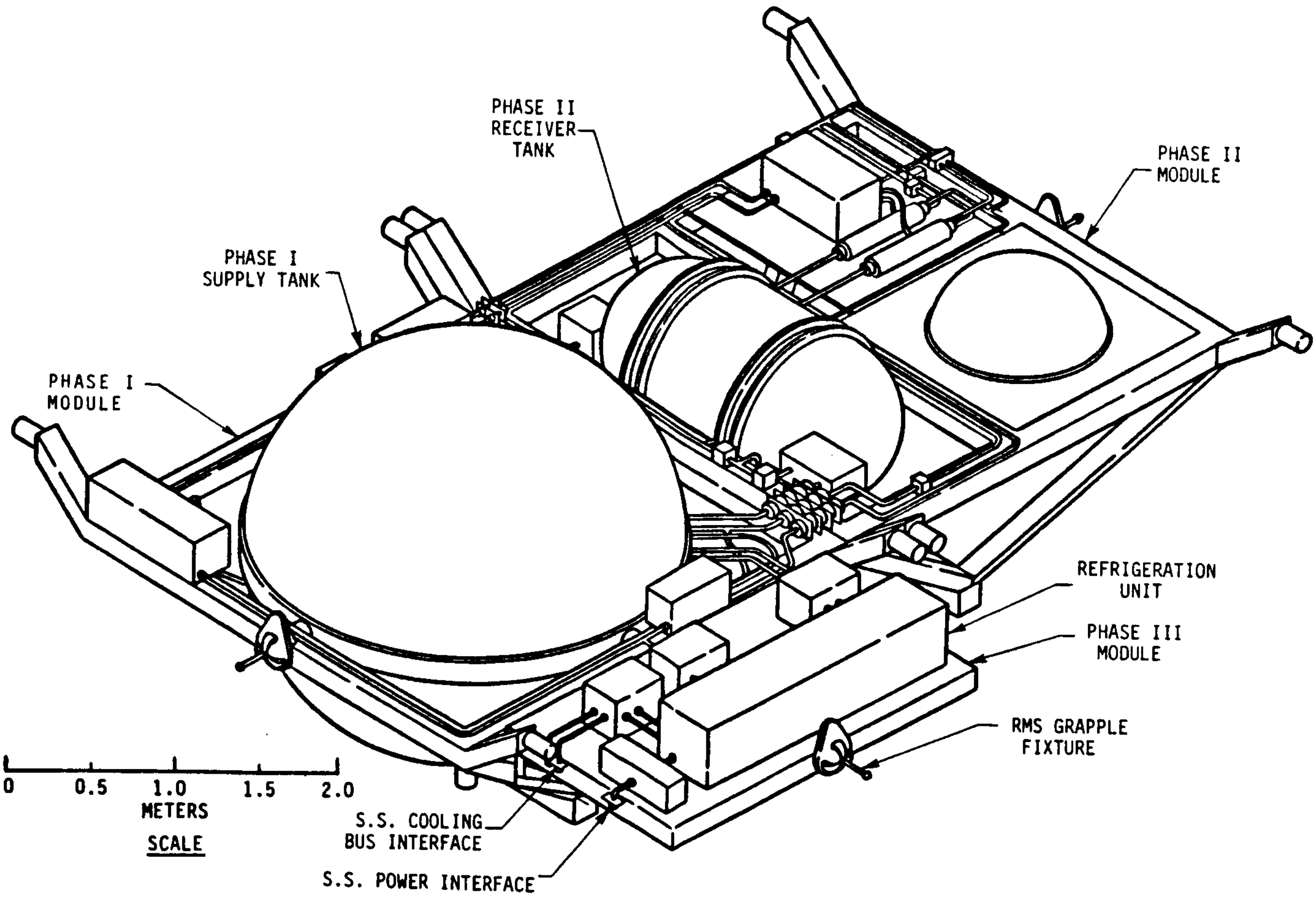
*6. Provide a sketch of the experiment including approximate dimensions.



PHASE I



PHASE II


PHASE II
RECEIVER
TANK
PHASE II
MODULE
PHASE I
SUPPLY TANK
PHASE I
MODULE
REFRIGERATION
UNIT
PHASE III
MODULE
RMS GRAPPLE
FIXTURE
0 0.5 1.0 1.5 2.0
METERS
SCALE
S.S. COOLING
BUS INTERFACE
S.S. POWER INTERFACE

PHASE III

7. Provide an equipment list including approximate dimension and weights as available.

EQUIPMENT	LENGTH cm (in)	WIDTH OR DIA cm (in)	HEIGHT cm (in)	WEIGHT (kg) (lbm)
Supply Dewar	267 (105)	267 (105)	--	900 (1985)*
Data Acquisition and Control System	76 (30)	30 (12)	51 (20)	100 (220)
Interface Panel	91 (36)	61 (24)	30 (12)	45 (99)
Support Structure	381 (150)	457 (180)	279 (110)	900 (1985)
<u>Additional Phase II Hardware</u>				
Receiver Tank	213 (84)	122 (48)	--	130 (287)
Pressurization System	152 (60)	122 (48)	122 (48)	100 (220)
Support Structure	234 (92)	457 (180)	203 (80)	200 (441)
<u>Additional Phase III Hardware</u>				
Refrigeration System	234 (92)	61 (24)	61 (24)	550 (1213)
Support Structure	234 (92)	61 (24)	30 (12)	250 (551)

* Including LH_2 mass

ORBIT CHARACTERISTICS

8. What properties of the orbit are especially important to your mission and why? (Plasma density, earth distance, etc.)

   Low-g, thermal environment, micrometeroid and atomic oxygen environment.

POINTING/ORIENTATION

9. Why have you chosen a particular view direction? Is it a requirement?

   No particular view direction is specified.

10. Is the experiment capable of providing self-orientation? Describe equipment and procedures.

    No self orientation is required for the LTCFS experiment.

11. If Space Station were oriented in a direction other than your desired orientation, how would your experiment be affected?

Solar	Change in solar orientation would change thermal environment of experiment, but would have no detrimental effects on experiment objectives.
Earth	Same as above.
Inertial	Same as above.

POWER

12. List components requiring electrical power and the desired operating levels of power and voltage.

ITEM	VOLTAGE (V)	POWER (W)	AC/DC Or N/A
Phase I - Data Acquisition and Control System	28V	100	DC
Phase II - Data Acquisition and Control System	28V	100/600*	DC
Phase III - Data Acquisition and Control System	28V	100	DC
Phase III - Active Refrigeration System	400V	2500	AC

* 100 w nominal/600 w peak

13. Could you use power distributed in the following conditioned forms? Why or why not?

YES	NO		CIRCLE PREFERENCE		
X		High Frequency AC	20 KHz	Other____	
X		Low Frequency AC	70 Hz	400 Hz	
X		DC	28V	120V	270V

14. What special power conditioning requirements do you have?

None.

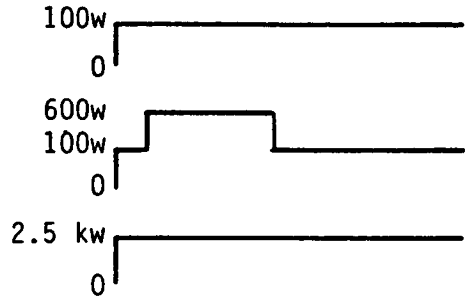
15. Sketch typical load profiles for power usage:

(a) Over an orbit (90 min)

PHASE I

PHASE II

PHASE III

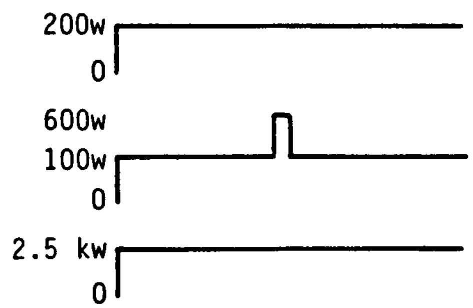


(b) Over a day

PHASE I

PHASE II

PHASE III

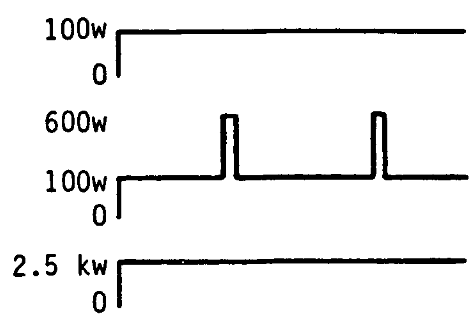


(c) Over a month

PHASE I

PHASE II

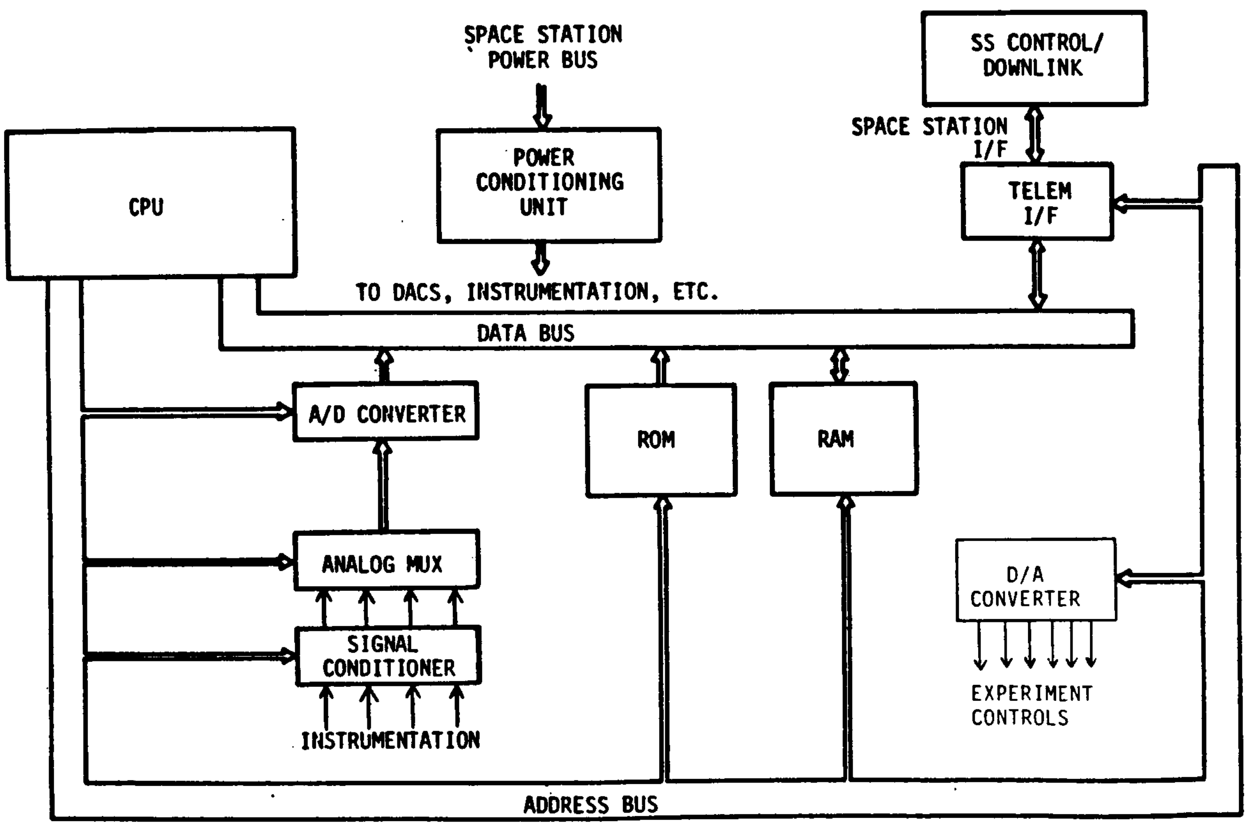
PHASE III



16. Describe standby operations. What are the consequences for an interruption of electrical power?

In standby mode, the experiment gathers thermal performance data at a rate of 10 samples per minute. Critical operational parameters, such as pressure and temperature limits, are monitored and controlled to ensure proper system operation. Interruption of electrical power would result in loss of experiment data. All controls critical to experiment safety, such as tank pressure control, are backed up with a passive single failure tolerant system. For example, a pressure relief valve in parallel with a burst disc will prevent hydrogen tank overpressures from occurring if the active control system fails.

17. Sketch the experiment electrical system in block diagram form.



BLOCK DIAGRAM - DATA ACQUISITION AND CONTROL SYSTEM

DATA COMMUNICATIONS

18. Describe what you want to do with the uplink and the downlink.

Uplink communications would be used for ground control and status inquiries on experiment. Downlink communications will be used for experiment data transmission to the ground and for replies to control and status inquiries.

19. Describe specific monitoring needs, both on-board and ground based (e.g. visual observation of deployment by Shuttle crew, monitoring of experiment performance parameters by ground crew, etc).

Normal experiment operation, with the exception of fluid transfer operations will be unattended. Data downlinks will occur approximately every 7 days and will require crew interaction during downlink times. Fluid transfer operations during phase II will require crew initiation and monitoring, approximately 2 times per month for 6 months. Experiment caution and warning system will notify crew of experiment anomolies.

20. Describe data transmission and storage needs, including the nature of the information and rationale for on-board storage.

Due to low data sampling rate, approximately 10 samples per hour, data may be stored on board Space Station and downlinked approximately every 7 days. Data will be primarily experiment measurements, such as pressure, temperature, etc. Data transmission and storage requirements by phase are as follows:

THERMAL

21. Identify major sources of heat and describe heat rejection provisions. (Include operating temperatures of specific components and estimated loads.)

    The major source of heat in Phase I and II is the Data Acquisition and Control System (DACS). The DACS will use passive and active thermal control independent of Space Station. Phase III has a 2.5kW heat dissipation at 300K from the active refrigeration system. The Space Station thermal bus system will be utilized for Phase III heat dissipation.

22. What special interfaces with Space Station will be required for adequate thermal control? (e.g. low temperature requirements, extremely uniform temperature, etc).

    No special interfaces required.

23. Identify problems that may occur with:

    (a) Overheating
    Over temperature and potential failure of temperature sensitive components, such as electronics.

    (b) Overcooling
    Under temperature and potential failure of temperature sensitive components, such as electronics.

EQUIPMENT PHYSICAL CHARACTERISTICS

24. Describe critical aspects of location on or within Space Station related to:

Viewing angle
None

G-level
None

Thermal Control
Location accessible to Space Station thermal bus is desired

Contamination
Experiment will vent approximately 8 kg (18 lbm) of GH_2 during each fluid transfer operation. It may be desirable to place experiment away from areas where such contamination is undesirable.

Accessibility
Location must be accessible for experiment mounting, reconfiguration and removal. Accessibility by the Mobile Remote Manipulator System (MRMS) is desired.

25. Describe requirements to go from shuttle stowed to Space Station operational (e.g. self-contained, self-deployed, located in lab module, etc).

Experiment requires EVA and/or MRMS operation for deployment. Major tasks during deployment are:

1. Mounting on Space Station structure
2. Connection of power and telemetry interfaces
3. Connection of thermal bus.

26. Describe Space Station integration requirements such as attachments, ports, supply lines and storage, etc.

The following interfaces are required:

1. Power:
   - o 100W to 600W - 28 VDC
   - o 2.5kW - 400 Hz, 400 VAC
2. Space Station Data Acquisition System Interface
3. Physical attachment to Space Station structure
4. Thermal bus interface (2.5kW capability desired)

27. If a remote location is desired, explain why.

There is no current requirement for a remote location. Safety and contamination issues need to be investigated to determine if they dictate a remote location. Current baselined experiment location is adjacent to proposed location for OTV servicing bay on growth station.

28. Describe special environmental requirements (e.g. pressurization, temperature, etc).

There are no special environmental requirements. Normal, on-orbit environment is adequate.

CREW REQUIREMENTS

29. Describe the nature of crew assignments during operation and standby.

    Crew will be required to downlink data to ground approximately every seven days. During Phase II, crew initiation and monitoring of fluid transfer operations will be required. The transfer operations will last for approximately 2 hours and will occur twice a month for about six months.

30. Describe specific tasks related to deployment and retrieval.

    1. Mounting and removal of hardware from Space Station structure.

    2. Mounting and removal of Space Station power, data and cooling interfaces.

31. Describe crew activities that would be performed on a routine basis for maintaining operational status.

    Downlink of experiment data approximately every 7 days.

SERVICING

32. Describe the nature of consumables and returnables desired and the frequency of resupply and return.

    Liquid hydrogen is consumed; however, no resupply or return of materials is required.

33. What special attachments or spatial allocations will be necessary for storage of consumables?

    None.

CONTAMINATION

34. What contaminants may be released by the experiments? (e.g. gaseous products, particulates, etc.)

    Experiment will vent 8k kg (18 lbm) of gaseous hydrogen during each fluid transfer operation.

35. What contaminants would detrimentally affect the experiment? (e.g. from thruster effluent, ECLS waste, etc.)

Contaminants will not detrimentally affect the experiment.

SAFETY

36. Are there any specific safety requirements or hazards?

Experiment will contain 6.5 m^3 (228 ft^3) of liquid hydrogen and will vent gaseous hydrogen during fluid transfer operations and any required contingency dump operations.

SPECIAL CONSIDERATIONS

37. Identify other unique features of the experiment that are important for Space Station design and operation.

None.

FLIGHT PRECURSORS

*38. Identify ground, shuttle, or other flight precursors which might be performed prior to Space Station implementation.

Experiment will be ground qualified for shuttle launch and orbital environment prior to Space Station implementation.

APPENDIX E

MRDB FORMS

MISSION DESIGN

MISSION CODE: TDMX2311

PAYLOAD ELEMENT NAME: LONG-TERM CRYOGENIC STORAGE

COUNTRY: UNITED STATES OF AMERICA

CONTACT: MR. ROGER SCARLOTTI
BEECH AIRCRAFT
P. O. BOX 9631
BOULDER, COLORADO 80301

PHONE: 303-443-1650

STATUS: 3

NASA Space Station Mission Data Base - Form 1

FLIGHTS:

	1991	1992	1993	1994	1995	1996	1997	1998	1999	2000
EQUIPMENT UP (flight)	0	0	1	0	1	1	0	0	0	0
EQUIPMENT DOWN (no. of times)	0	0	0	0	0	0	1	0	0	0
OPERATIONAL DAYS (per flight)	0	0	365	365	365	365	90	0	0	0
OTV FLIGHTS	0	0	0	0	0	0	0	0	0	0

EARLY FLIGHT: --

LATE RETURN: --

OBJECTIVE:

LINE:

1 THE EXPERIMENT WILL DEMONSTRATE LONG-TERM STORAGE AND TRANSFER OF
2 CRYOGENIC FLUIDS IN AN ON-ORBIT ENVIRONMENT. VARIOUS TECHNOLOGIES
3 UTILIZED IN LONG-TERM STORAGE AND TRANSFER OF CRYOGENS WILL BE TESTED
4 AND EVALUATED. THESE TECHNOLOGIES RANGE FROM BASIC PASSIVE TECHNOLOGIES
5 SUCH AS THICK MULTI-LAYER INSULATION BLANKETS TO ACTIVE REFRIGERATION
6 SYSTEMS.

DESCRIPTION:

LINE:

1 THE EXPERIMENT CONSISTS OF THREE PHASES. PHASE I WILL TEST BASIC PASSIVE
2 TECHNOLOGIES UTILIZED IN LONG-TERM CRYOGENIC STORAGE. A 6.5 CUBIC METER (228
3 CUBIC FOOT) TANK WILL BE MOUNTED TO SPACE STATION, AND THERMAL PERFORMANCE
4 DATA WILL BE MEASURED FOR A PERIOD OF TWO YEARS. PHASE II WILL DEMONSTRATE
5 CRYOGENIC FLUID TRANSFER TECHNOLOGIES. THE PHASE I HARDWARE WILL BE
6 RECONFIGURED ON-ORBIT BY ADDING A RECEIVER TANK AND OTHER NECESSARY
7 HARDWARE. PHASE II EXPERI-MENTATION WILL BE PERFORMED FOR A PERIOD OF ONE
8 YEAR. PHASE III WILL DEMONSTRATE ACTIVE REFRIGERATION TECHNOLOGY. A PALLET
9 CONTAINING AN ACTIVE REFRIGERATION SYSTEM WILL BE FLOWN TO SPACE STATION AND
10 ATTACHED TO THE PHASE II HARDWARE. THE REFRIGERATION SYSTEM WILL BE TESTED FOR
11 A PERIOD OF AT LEAST ONE YEAR.

TYPE NUMBER: 16
IMPORTANCE OF SPACE STATION: 7
NON-SERVICING OMV FLIGHT (per year): 0

NASA Space Station Mission Data Base - Form I (concluded)

ORBIT

MISSION CODE: TDMX2311

ORBIT: I (If I is selected, skip remainder of Form 2)

APOGEE: ______________ km + ________ km
- ______________ km TOLERANCE

PERIGEE: ______________ km + ______________ km
- ______________ km TOLERANCE

INCLINATION: ______________ km + ______________ km
- ______________ km TOLERANCE

LOCAL TIME OF EQUATOR CROSSING NODE: ________ hr ________ min
ASCENDING OR DESCENDING: ________

SPECIAL CONSIDERATIONS (ORBIT):

LINE
1
2
3
4

POINTING/ORIENTATION

MISSION CODE: TDMX2311

POINTING/ORIENTATION: 1 (If 1 is selected, skip remainder of FORM 3)

VIEW DIRECTION:

If 4 selected, OTHER:

HOURS:

TRUTH SITES:

POINTING ACCURACY:	sec
POINTING KNOWLEDGE:	sec
FIELD OF VIEW:	deg
POINTING STABILITY RATE:	sec per sec
POINTING STABILITY:	sec
PLACEMENT:	sec

SPECIAL CONSIDERATIONS

LINE
1
2
3
4

POWER

MISSION CODE: TDM2311

POWER: 5 kw

	AC	DC
OPERATING (kw):	2.5	.10
HOURS, PER DAY (operating)	24	24
VOLTAGE:	400	28
FREQUENCY:	400 Hz	
PEAK (kW):	--	--
HOURS PER DAY (peak):	--	--
STANDBY POWER (kw):	--	--

SPECIAL CONSIDERATIONS (power):

LINE

THE ABOVE POWER REQUIREMENT (2.5 kw) IS REQUIRED FOR THE THIRD FLIGHT ONLY. PHASE ONE AND TWO POWER REQUIREMENTS ARE 100 WATTS, WITH PEAKS OF 600 WATTS DURING PHASE II TRANSFER OPERATIONS. (PEAK DURATION: 60MIN/TRANSFER

THERMAL

MISSION CODE: TDMX2311

THERMAL: 3

	ACTIVE		PASSIVE	
	OPER	NON-OPER	OPER	NON-OPER
MIN TEMP (°C)	0	-100	0	-100
MAX TEMP (°C)	50	100	100	100
MIN HEAT REJECTION (KW)	0	0	.1	0
MAX HEAT REJECTION (KW)	2.5	0	.6	0

SPECIAL CONSIDERATIONS:

LINE

1 ACTIVE COOLING OF 2.5 KW IS REQUIRED ONLY DURING THE THIRD PHASE. IF ADEQUATE

2 COOLING CAPACITY IS NOT AVAILABLE, A DEDICATED RADIATOR SYSTEM IS A VIABLE

3 ALTERNATIVE TO USE OF THE SPACE STATION THERMAL BUS.

DATA/COMMUNICATIONS

MISSION CODE: TDMX2311

ON-BOARD DATA PROCESSING REQUIRED: 1

If 1, (YES), this DESCRIPTION:
ON-BOARD PROCESSING REQUIRED FOR HEALTH AND SAFETY STATUS OF EXPERIMENT.

ON-BOARD STORAGE (MBIT): 2

STATION DATA REQUIRED:

LINE

1 NONE

COMMUNICATION LINKS:

1.	From: Station To: Ground	DIGITAL DATA	VIDEO DATA	VOICE
a.	Generation rate (kbps)	1.0		NA
b.	Duration (hours)	0.3		0.5
c.	Frequency (per day)	0.1		0.1
d.	Delivery time (hours)	48		0
e.	Security (yes/no)	NO		NO
f.	Reliability (%)	99%		99%
g.	Interactive (yes/no)	NO		YES

2.	From: Ground To: Station	DIGITAL DATA	VIDEO DATA	VOICE
a.	Generation rate (kbps)			NA
b.	Duration (hours)			0.1
c.	Frequency (per day)			0.1
d.	Delivery time (hours)			0
e.	Security (yes/no)			NO
f.	Reliability (%)			99%
g.	Interactive (yes/no)			YES

3. From: Station To: Free Flyer	DIGITAL DATA	VIDEO DATA	VOICE
a. Generation rate (kbps)			NA
b. Duration (hours)			
c. Frequency (per day)			
d. Delivery time (hours)			0
e. Security (yes/no)			
f. Reliability (%)			
g. Interactive (yes/no)			YES

4. From: Free Flyer To: Station	DIGITAL DATA	VIDEO DATA	VOICE
a. Generation rate (kbps)			NA
b. Duration (hours)			
c. Frequency (per day)			
d. Delivery time (hours)			0
e. Security (yes/no)			
f. Reliability (%)			
g. Interactive (yes/no)			YES

5. From: Station To: Platform	DIGITAL DATA	VIDEO DATA	VOICE
a. Generation rate (kbps)			NA
b. Duration (hours)			
c. Frequency (per day)			
d. Delivery time (hours)			0
e. Security (yes/no)			
f. Reliability (%)			
g. Interactive (yes/no)			YES

6. From: Platform To: Station	DIGITAL DATA	VIDEO DATA	VOICE
a. Generation rate (kbps)	1.0		NA
b. Duration (hours)	0.1		
c. Frequency (per day)	0.1		
d. Delivery time (hours)	1		0
e. Security (yes/no)	NO		
f. Reliability (%)	99%		
g. Interactive (yes/no)	NO		YES

NASA Space Station Mission Data Base - Form 6 (continued)

7.	From: Platform To: Ground	DIGITAL DATA	VIDEO DATA	VOICE
a.	Generation rate (kbps)			NA
b.	Duration (hours)			
c.	Frequency (per day)			
d.	Delivery time (hours)			0
e.	Security (yes/no)			
f.	Reliability (%)			
g.	Interactive (yes/no)			YES

8.	From: Ground To: Platform	DIGITAL DATA	VIDEO DATA	VOICE
a.	Generation rate (kbps)			NA
b.	Duration (hours)			
c.	Frequency (per day)			
d.	Delivery time (hours)			0
e.	Security (yes/no)			
f.	Reliability (%)			
g.	Interactive (yes/no)			YES

9.	From: Station To: Shuttle	DIGITAL DATA	VIDEO DATA	VOICE
a.	Generation rate (kbps)			NA
b.	Duration (hours)			
c.	Frequency (per day)			
d.	Delivery time (hours)			0
e.	Security (yes/no)			
f.	Reliability (%)			
g.	Interactive (yes/no)			YES

10.	From: Shuttle To: Station	DIGITAL DATA	VIDEO DATA	VOICE
a.	Generation rate (kbps)			NA
b.	Duration (hours)			
c.	Frequency (per day)			
d.	Delivery time (hours)			0
e.	Security (yes/no)			
f.	Reliability (%)			
g.	Interactive (yes/no)			YES

NASA Space Station Mission Data Base - Form 6 (continued)

COMMENTS:

LINE

1 DATA WILL BE DOWNLINKED APPROXIMATELY EVERY SEVEN DAYS. DOWNLINK TIME

2 AND FREQUENCY IS NOT CRITICAL.

NASA Space Station Mission Data Base - Form 6 (concluded)

EQUIPMENT

MISSION CODE: TDMX2311

MODULE CODE: 1

SHARED FACILITY CODE: 0
(If 1 is selected, list mission codes of sharing missions below:)

EQUIPMENT LOCATION:

Equipment location is:	3 EXTERNAL/ATTACHED UNPRESSURIZED
Dimensions (m)	
Length	3.0
Width	3.5
Height	3.0
Volume (m^3)	31.5
Pkg Dimension (m)	
Length	3.0
Width	3.5
Height	3.0
Pkg Volume (m^3)	31.5
Launch Mass (kg)	1945.0
Acceleration Mass (g)	10^{-3}

ATTACH POINTS: 1
SET UP CODE: 1 2

HARDWARE DESCRIPTION:

LINE

THE ABOVE DIMENSIONS ARE FOR PHASE I HARDWARE ONLY. PHASE I CONSISTS OF A 6.5 CUBIC METER LIQUID HYDROGEN TANK WITH DATA ACQUISITION AND CONTROL HARDWARE AND SUPPORT STRUCTURE. PHASE II AND III RECONFIGURATIONS ARE DESCRIBED ON FORM 10. CURRENT BASELINED LOCATION IS ADJACENT TO THE PROPOSED LOCATION FOR THE OTV SERVICING BAY ON GROWTH STATION.

NASA Space Station Mission Data Base - Form 7 (concluded)

CREW

MISSION CODE: TDMX2311

INITIAL CONSTRUCTION/SET UP: 1 (If 9, skip to DAILY OPERATIONS)

TASK:

INTERFACE EXPERIMENT WITH SPACE STATION POWER AND DATA BUSSES.

PERIOD: 3 DAYS

IVA TOTAL CREW TIME: 24 MAN-HRS

EVA PRODUCTIVE CREW TIME: 12 MAN-HRS

SKILLS: (See last page of Form 8 for example)

Enter number of skill type/levels required:

SKILL TYPE

SKILL LEVEL	1	2	3	4	5	6	7
1							2
2							
3							

DAILY OPERATIONS: 0 (If 0, skip to PERIODIC OPERATIONS)

TASK:

IVA CREW TIME PER DAY: ______ MAN-HRS

SKILLS:

Enter number of skill type/levels required:

PERIODIC OPERATIONS: 1 (If 0, skip to TEARDOWN AND STOW)

TASK:

DOWNLINK DATA TO GROUND AND STATUS EXPERIMENT OPERATION.

IVA OCCURRENCE INTERVAL: 7 DAYS

CREW TIME/OCCURRENCE: 1 MAN-HRS

EVA OCCURRENCE INTERVAL: 0 DAYS

PRODUCTIVE CREW TIME/OCCURRENCES: 0 MAN-HRS

SKILLS:

Enter number of skill type/levels required:

SKILL LEVEL	SKILL TYPE 1	2	3	4	5	6	7
1							1
2							
3							

TEARDOWN AND STOW: 1 (If 0, skip this section)

TASK:

REMOVE SPACE STATION INTERFACES, SAFE EXPERIMENT FOR STORAGE IN SHUTTLE.

PERIOD: 1 DAY

IVA TOTAL CREW TIME: 12 MAN-HRS

EVA PRODUCTIVE CREW TIME: 12 MAN-HRS

SKILLS:

Enter number of skill type/levels required:

SKILL LEVEL	SKILL TYPE 1	2	3	4	5	6	7
1							2
2							
3							

NASA Space Station Mission Data Base - Form 8 (continued)

COMMENTS:

LINE

1

2

3

4

TYPICAL EXAMPLE OF SKILL TYPE/LEVEL MATRIX INPUT:

Skill Types

1. No Special Skill Required
2. Medical/Biological
3. Physical Sciences
4. Earth and Ocean Sciences
5. Engineering
6. Astronomy
7. Spacecraft Systems

Skill Levels

1. Task Trainable
2. Technical
3. Professional

If two medical/biological professionals are required, put 2 in second column, third row.

No more than 6 skill types can be used for a given task.

SERVICING

MISSION CODE: TDMX2311
SERVICING: 1 (If 1 is selected, skip remainder of Form 9)
SERVICE INTERVAL (DAYS):
CONSUMABLES
TYPES:

WEIGHT: kg
RETURN: kg
VOLUME UP: m^3
VOLUME DOWN: m^3
POWER: kw
HOURS FOR POWER: hrs
EVA HOURS PER SERVICE: hrs
TYPICAL TASKS (EVA):

IVA HOURS PER SERVICE: hrs
LOCATION OF SERVICING:
TYPICAL TASKS (IVA):

SPECIAL CONSIDERATIONS:
LINE
1
2
3

CONFIGURATION CHANGES

MISSION CODE: TDMX2311

CONFIGURATION CHANGES: 2 (If 1 selected, skip the remainder of Form 10)

INTERVAL: 500 days (average)

CHANGE-OUT EQUIPMENT

TYPE: ADD RECEIVER TANK AND PRESSURIZATION SYSTEM FOR PHASE II TESTING

ADD ACTIVE REFRIGERATION MODULE TO TEST HARDWARE FOR PHASE III TESTING

	PHASE II		PHASE III	
WEIGHT:	430	kg	800	kg
RETURN:	0	kg	0	kg
VOLUME UP:	22	m^3	0.5	m^3
VOLUME DOWN:	0	m^3	0	m^3
POWER:	0	kw	0	kw
HOURS FOR POWER:	0	hrs	0	hrs
EVA HOURS PER CHANGE:	12	hrs	12	hrs

TYPICAL TASKS (EVA): INTERFACE TRANSFER LINES AND PRESSURIZATION SYSTEM TO PHASE I CONFIGURATION

INTERFACE ACTIVE REFRIGERATION SYSTEM TO PHASE II CONFIGURATION.

IVA HOURS PER CHANGE: 24 hrs

LOCATION: I

TYPICAL TASKS (IVA):

SUPPORT OF EVA AND CHECKOUT OF NEW CONFIGURATION.

SPECIAL CONFIGURATIONS:

LINE

1 WEIGHT AND VOLUME TAKEN DOWN AFTER COMPLETION OF PHASE IV ARE 2750

2 kg AND 54 m^3 RESPECTIVELY. TOTAL MASS DOWN DOES NOT INCLUDE H_2.

3

SPECIAL NOTES

MISSION CODE: TDMX2311

CONTAMINATION:

LINE

1 DURING EACH OF THE TEN (10) TRANSFER OPERATIONS PERFORMED DURING PHASE II,

2 THE EXPERIMENT WILL VENT APPROXIMATELY 18 kg (18 lbm) OF HYDROGEN OVER A PERIOD

3 OF 1-2 HOURS.

STRUCTURES:

LINE

1

2

MATERIALS:

LINE

1

2

RADIATION:

LINE

1

2

NASA Space Station Mission Data Base - Form 11

SAFETY:

LINE

1 EXPERIMENT WILL CONTAIN A 6.5 CUBIC METER (228 CUBIC FOOT) LIQUID HYDROGE

2 DEWAR.

STORAGE:

LINE

1

2

OPTICAL WINDOW:

LINE

1

2

SCIENTIFIC AIRLOCK:

LINE

1

2

TETHER:

LINE

1

2

VACUUM VENTING:

LINE

1 EXPERIMENT WILL VENT 8 kg (18 lbm) OF HYDROGEN OVER A PERIOD OF 1-2

2 HOURS DURING EACH OF TEN FLUID TRANSFER OPERATIONS. TRANSFER

3 OPERATIONS WILL OCCUR APPROXIMATELY TWICE A MONTH.

OTHER:

LINE

1

2

3

4

NASA Space Station Mission Data Base - Form 11 (concluded)

APPENDIX F

ABBREVIATIONS AND SYMBOLS

ABBREVIATIONS AND SYMBOLS

deg	Degree
ft	Feet
ft^3	Cubic Feet
hr	Hour
in	Inches
kbps	One Thousand Bits Per Second
kPa	Kilopascal
kw	Kilowatts
lbm	Pound-mass
low-g	Low Gravity
m	Mass
m^3	Cubic Meters
min	Minute
mm	Millimeter
psia	Pounds per Square Inch
w	Watts
A	Area
A/D	Analog to Digital
AC	Alternating Current
AFWAL	Air Force Wright Aeronautical Laboratories
ARC	Ames Research Center
AFRPL	Air Force Rocket Propulsion Lab
ATS	Applications Technology Satellite
Ag	Silver
Al	Aluminum
B/O	Boiloff
BOL	Beginning-of-Life
Btu	British Thermal Unit
CDR	Critical Design Review
CFMFE	Cryogenic Fluid Management Flight Experiment
CIU	Coolant Interface Unit
CPU	Central Processing Unit
Cres	Stainless Steel

ABBREVIATIONS AND SYMBOLS

(continued)

D/A	Digital to Analog
DACS	Data Acquisition and Control System
DC	Direct Current
DDT&E	Design, Development Test and Engineering
DMS	Data Management System
DOD	Department of Defense
ECLS	Environmental Control and Life Support
ELMS	Earth Limb Measurement Satellite
EOL	End-of-Life
ET	External Tank
EVA	Extra-Vehicular Activity
FCSS	Fuel Cell Servicing System
FEP	Fluorinated Ethylene Propylene
GEO	Geosynchronous Earth Orbit
GH2	Gaseous Hydrogen
GO_2	Gaseous Oxygen
GRMS	Root Mean Square G-Level
GSE	Ground Support Equipment
GSFC	Goddard Space Flight Center
HEX	Heat Exchanger
HPG	High Pressure Gas
HTTA	Hydrogen Thermal Test Article
Hz	Hertz
I/F	Interface
IOC	Initial Operational Capability
IR	Infra-red
IRAS	Infra-red Astronomical Observatory
IVA	Intra-Vehicular Activity
J	Joule
JPL	Jet Propulsion Laboratory
JSC	Johnson Space Center
J-T	Joule-Thomson
K	Kelvin
KHz	Kilohertz

ABBREVIATIONS AND SYMBOLS
(continued)

KSC	Kennedy Space Center
Kg	Kilogram
Km	Kilometer
L	Liter
LAD	Liquid Acquisition Device
LDEF	Long Duration Exposure Facility
LEO	Low Earth Orbit
LH_2	Liquid Hydrogen
LHe	Liquid Helium
LN_2	Liquid Nitrogen
LOX	Liquid Oxygen
LRC	Langley Research Center
LTCFSE	Long-Term Cryogenic Fluid Storage Experiment
LaNi5	Lanthanum Nickel
LeRC	Lewis Research Center
MBIT	One Million Bits
MLI	Multi-Layer Insulation
MMU	Manned Maneuvering Unit
MRDB	Mission Requirements Data Base
MRMS	Mobile Remote Manipulator System
MSFC	Marshall Space Flight Center
MUX	Multiplexer
NASA	National Aeronautics and Space Administration
OAO	Orbiting Astronomical Observatory
OD	Outside Diameter
OMS	Orbital Maneuvering System
OS	Outer Shell
OSR	Optical Solar Reflector
OTTA	Oxygen Thermal Test Article
OTV	Orbital Transfer Vehicle
PCU	Power Conditioning Unit

ABBREVIATIONS AND SYMBOLS
(continued)

PDR	Preliminary Design Review
PODS	Passive Orbital Disconnect Strut
PRSA	Power Reactant Supply Assembly
PSD	Power Spectral Density
PSR	Program Safety Review
PV	Pressure Vessel
Q	Heat Flow (Heat Leak)
RAM	Random Access Memory
RF	Radio Frequency
RITS	Rod in Tube Support
RMS	Remote Manipulator System
ROM	Rough Order of Magnitude
RTG	Radioisotope Thermoelectric Generator
RTLS	Return to Launch Site
SEC	Second
SIRTF	Space Infra-Red Telescope Facility
SOA	State-of-the-Art
SS	Space Station
STS	Space Transportation System
SfHe	Superfluid Helium
TDAG	Technology Development Advocacy Group
TDM	Technology Development Mission
TORF	Tethered Orbital Refueling Facility
TDRS	Tracking and Data Relay Satellite
TNKCAP	Tank Cooldown Analysis Program
TVS	Thermodynamic Vent System
UARS	Upper Atmosphere Research Satellite
V	Volts
V	Volume
VCS	Vapor Cooled Shield
WBS	Work Breakdown Structure

ABBREVIATIONS AND SYMBOLS
(concluded)

^{o}R	Degrees Rankine
α	Solar Absorptance
ϵ	Infra-red Emittance
σ	Stefan - Boltz mann constant ($5.6696\ 10^{-8} w/m^2K^4$)

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16. Abstract

This study presents the conceptual design of a space station Technology Development Mission (TDM) experiment to demonstrate and evaluate cryogenic fluid storage and transfer technologies. The experiment will be deployed on the IOC space station for a four-year duration. It is modular in design, consisting of three phases to test the following technologies:

Phase I - Passive Thermal Technologies
Phase II - Fluid Transfer Technologies
Phase III - Active Refrigeration Technologies

Use of existing hardware was a primary consideration throughout the design effort, resulting in recommendations to use several pieces of existing hardware in the experiment. A conceptual design of the experiment was completed, including configuration sketches, system schematics, equipment specifications, and space station resources and interface requirements. These requirements were entered into the NASA Space Station Mission Data Base. A program plan was developed defining a twelve-year development and flight plan at a cost of $94.3M (in 1986 dollars).

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